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U. S. ARMY
TRANSPORTATION RESEARCH COMMAND
FORT EUSTIS, VIRGINIA

TREC TECHNICAL REPORT 62-102

PRELIMINARY DESIGN STUDY
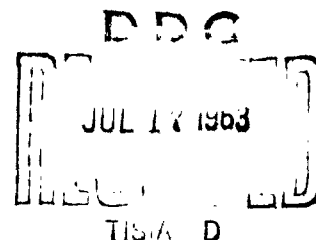## HOT CYCLE RESEARCH AIRCRAFT

SUMMARY REPORT

Task 1D121401D14403
(Formerly Task 9X38-01-020-03)

Contract DA 44-177-TC-832

March 1963

prepared by:

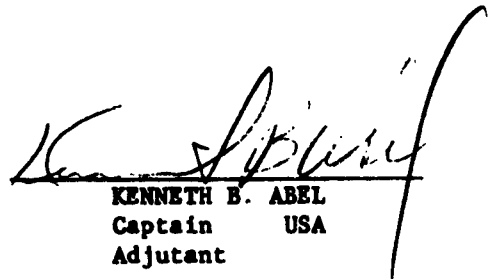HUGHES TOOL COMPANY
Aircraft Division
Culver City, California

Under the terms of Contract DA 44-177-TC-832, Hughes Tool Company, Aircraft

Division, has conducted a preliminary design study of a hot cycle propulsion

system research aircraft utilizing the rotor system fabricated under Air

Force Contract AF 33(600)-30271. This study is in conjunction with the

efforts of the U. S. Army to improve the performance of rotary-wing aircraft.

The conclusions presented in this report are concurred in by the U. S.

Army Transportation Research Command, Fort Eustis, Virginia, the cognizant

agency for the contract.

FOR THE COMMANDER:

KENNETH B. ABEL
Captain   USA
Adjutant

APPROVED BY:

H. S. JOHNSON
USATRECOM Project Engineer

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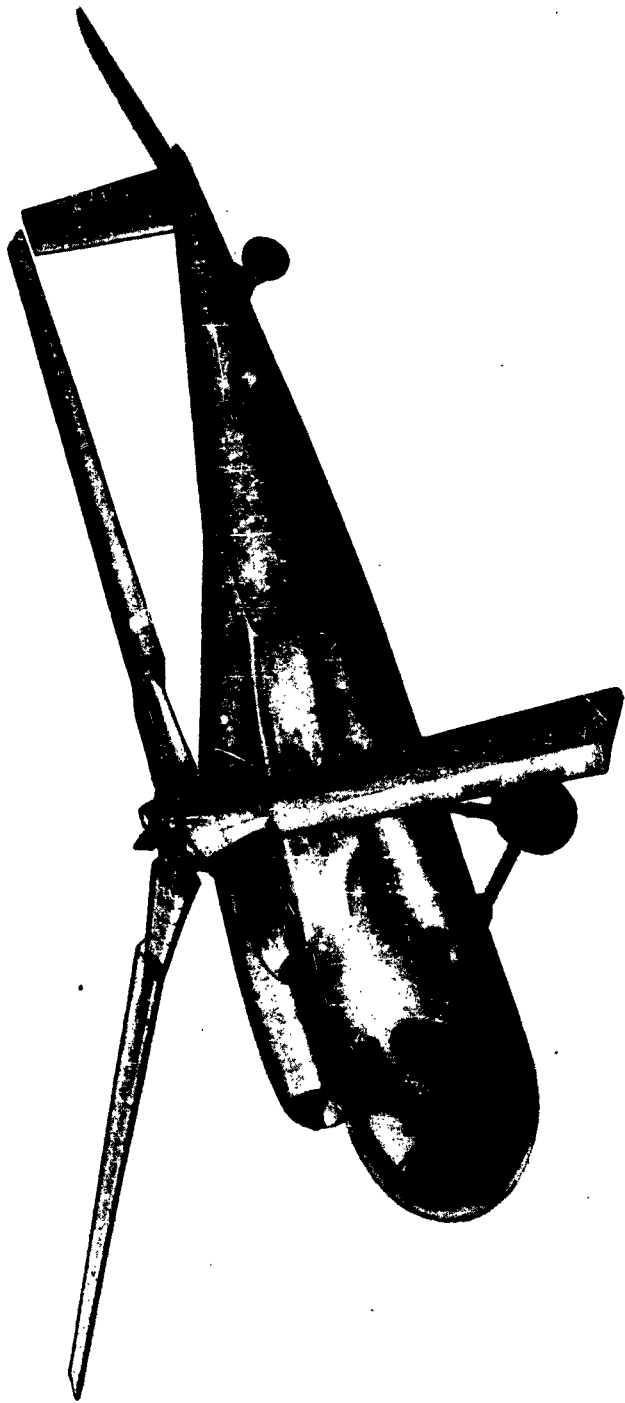SUMMARY REPORT

Report No. HTC-AD-62-31

Prepared by
Hughes Tool Company, Aircraft Division
Culver City, California

for
U.S. ARMY TRANSPORTATION RESEARCH COMMAND
FORT EUSTIS, VIRGINIA

ACKNOWLEDGEMENTS

FOREWORD

This report was prepared in accordance with paragraph 6, Clause 2 of Army Contract DA 44-177-TC-832 and summarizes the results of a preliminary design of a hot cycle propulsion system research aircraft performed under paragraph a, Clause 1 of the same contract. It also includes a preliminary design of the modifications that will be required to adapt an existing whirl test facility at the Contractor's plant to the twin-engine power plant installation of the research aircraft. This whirl test facility, fabricated under Contract AF 33(600)-30271, was used to perform a 60-hour feasibility whirl test on an existing rotor that will be installed on the research aircraft.

This work was performed at the Hughes Tool Company - Aircraft Division, Culver City, California, under the direction of Mr. H. O. Nay, Manager, Transport Helicopter Department, and under the direct supervision of Mr. J. L. Velazquez, Senior Project Engineer, Hot Cycle Program.

The Contract was effective on December 29, 1961. The work was completed on June 22, 1962.

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ILLUSTRATIONS

ILLUSTRATIONS (continued)

1. SUMMARY

In accordance with Army Transportation Research Command Contract DA 44-177-TC-832, a preliminary design study has been made of a hot cycle propulsion system research aircraft. This design was based on using the hot cycle pressure jet rotor system developed under Air Force Contract AF 33(600)-30271 and will incorporate two gas generator versions of the General Electric T64 engine. The design gross weight of the research aircraft will be 15,300 pounds at normal load factors. This preliminary design study included investigation of the general configuration, performance, weight and balance, stability and control, structural characteristics, and dynamic characteristics of the research aircraft.

The configuration of the aircraft chosen in this study permits operation as an autogyro as well as a helicopter. When operating as an autogyro at a weight sufficient for test purposes (10,000 pounds), this research aircraft is estimated to have a maximum speed of approximately 200 knots. The alternate overload gross weight will provide the capability of lifting a 15,000-pound payload in hover.

Based upon the configuration of the Research Aircraft as obtained from this study, a preliminary design was made of the modifications to the existing whirl tower that are required to permit testing of the complete rotor and propulsion system of the research aircraft. The study of modifications to the whirl tower is included here as an Annex to this report.

2. INTRODUCTION

This report summarizes the work accomplished under Army Contract No. DA 44-177-TC-832. The principal objective of that Contract was to perform a preliminary design study of a hot cycle propulsion system research aircraft utilizing the hot cycle pressure jet rotor system developed under Air Force Contract AF 33(600)-30271 and incorporating two T64 gas generators. The design gross weight of the aircraft was to be at least 15,000 pounds at normal load factors. The preliminary design was to establish the general configuration, dynamic characteristics, performance, stability and control, and structural characteristics of the research aircraft. The design study was to be based on the utilization of available components to the maximum extent possible. Information sufficient to establish the weight of all major systems as well as to permit evaluation of their functional and structural integrity was to be determined. Specification MIL-H-8501A was to be used as the stability and control objective.

This contract also required that the contractor furnish a preliminary design of the modifications to the whirl tower as required to permit testing of the complete rotor and propulsion system of the research aircraft. The whirl tower modifications are included as an Annex to this report.

The initial work done under this contract was an evaluation of a "flying crane" type of helicopter with a tubular steel fuselage. Earlier company-funded studies had indicated that this type of heavy-lift helicopter might be the best vehicle to demonstrate the flight characteristics of the hot cycle propulsion system. However, it became clear as this present study progressed that it was possible, with little, if any, increase in weight or cost, to arrange available major components such as the rotor, engines, diverter valves, and landing gear into the configuration shown on the frontispiece of this report, which is aerodynamically cleaner than the usual conception of a "flying crane." With this final configuration, it was found possible to obtain not only heavy-lift capability in hover, but very attractive cruise performance as well, with potential speeds of up to 200 knots in the autogyro mode. This report describes the evolution of the final chosen configuration and includes information on the estimated performance of the aircraft as well as a discussion of all major systems.

3. GENERAL CONFIGURATION STUDY

3.1 PRELIMINARY INVESTIGATION

Shortly before the present contract was started, the contractor made brief preliminary studies of possible configurations of research aircraft to demonstrate the suitability of the hot cycle propulsion system for helicopter applications. It was decided that a heavy-lift, "flying crane" aircraft, using existing components, would serve the purpose. The external cargo configuration, shown in Figure 1, was chosen with the thought of minimizing fuselage structure. Because of the drag of the external cargo, cruise speed with external payload was estimated to be about 80 knots. At this low cruise speed, it was felt that one engine could be placed forward and the other aft of the rotor center line without having serious differences in ram recovery pressure to the two engines. The fore and aft engine location shown would make easier the problem of balancing the helicopter. This first configuration was based on use of the existing hot cycle rotor, a modified Hughes 269A cabin, and an H-34 landing gear. The diverter valves (or dump valves as called out in Figure 1) were to be supplied by Hughes especially for this aircraft.

3.2 MISSIONS FOR RESEARCH AIRCRAFT

As soon as active design work started on the present contract, it became clear that an aircraft of the configuration of Figure 1 would have limited performance and utility for test purposes compared to what could be achieved with judicious rearrangement of the major components, plus reasonable attention to drag reduction. To permit a more complete evaluation of the concept, a change of viewpoint was caused by a realization of the design freedom and performance gains permitted for the basic helicopter. Also, test of the research aircraft in the autogyro mode could be conducted if the already available General Electric J85 diverter valve were to be used instead of the proposed special T64 diverter (dump) valve referred to in Section 3.1. A discussion of the features of the J85 valve is given in Section 3.4. Once this increased performance capability was recognized, the following tentative set of mission requirements was set down to be used as objectives of a test program for the research aircraft:

a. Investigation of rotor characteristics in free hovering flight and forward flight as a helicopter for loads, performance, and flying qualities.

b. Demonstration of heavy-lift capability in hover.

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c. Limited exploration of high-speed, unloaded-rotor auto-gyro mode flight regime.

3.3   OVER-ALL CONFIGURATION CONSIDERATIONS FOR PRELIMINARY DESIGN

In order to select the configuration best suited to evaluate the concept, the following criteria were used as design guides.

3.3.1   Maximum Net Lifting Ability

a. Minimum download on tail and fuselage

b. Minimum length of hot gas ducting to reduce pressure drops

c. Maximum engine inlet ram recovery and minimum distortion to conserve power available.

3.3.2   Weight and Strength

a. Efficient arrangement of structure to minimize weight in joints

b. Optimum fuselage depth for given fuselage length to obtain lowest weight fuselage

c. Landing gear location to provide adequate wheel base, tread, proper strut operation, and minimum carry-through structure.

3.3.3   Longitudinal Balance

Center of gravity to be located at centerline of rotor.

3.3.4   Satisfactory Flying Qualities (in addition to MIL-H-8501A)

a. Smooth transition into and from forward flight

b. Minimum shake and vibration due to fluctuating tail load

c. Adequate control power versus vertical CG location.

3.3.5   Cost and Schedule Requirements

a. Use of available components

b. Use of simple tooling

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H-34 LANDING GEAR

DUMP VALVE

ROTOR BLADE

WESTON TANDEM POWER CYLINDER 13971

145 IN.

MODIFIED 269A
CABIN

71 IN.

144 IN.

115 IN.

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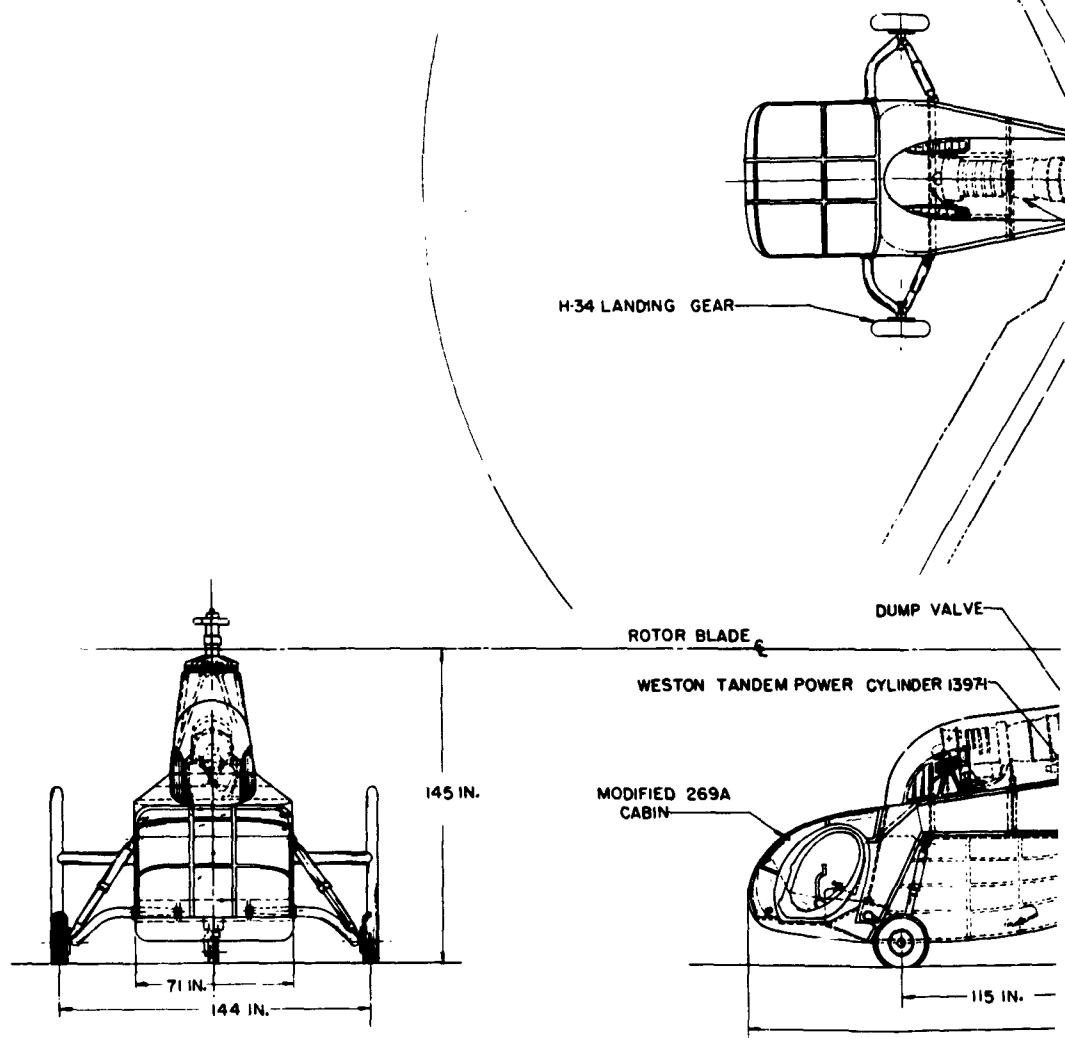
Figure 1. Initial Crane-Type Research Aircraft.

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2 G.E. T-64 GAS GENERATORS

145 IN.

HUB

HOT CYCLE ROTOR SYSTEM
(AVAILABLE FROM CONTRACT-AF 33(600)-30271)

DUMP VALVE

CYLINDER (397)

27.5 FT. RADIUS

YAW CONTROL

FUEL CELLS

BALLAST
LATCH

60 IN.

68 IN.

H-34 TAIL WHEEL

115 IN.

182 IN.

48 IN.

41 IN.

463 IN.
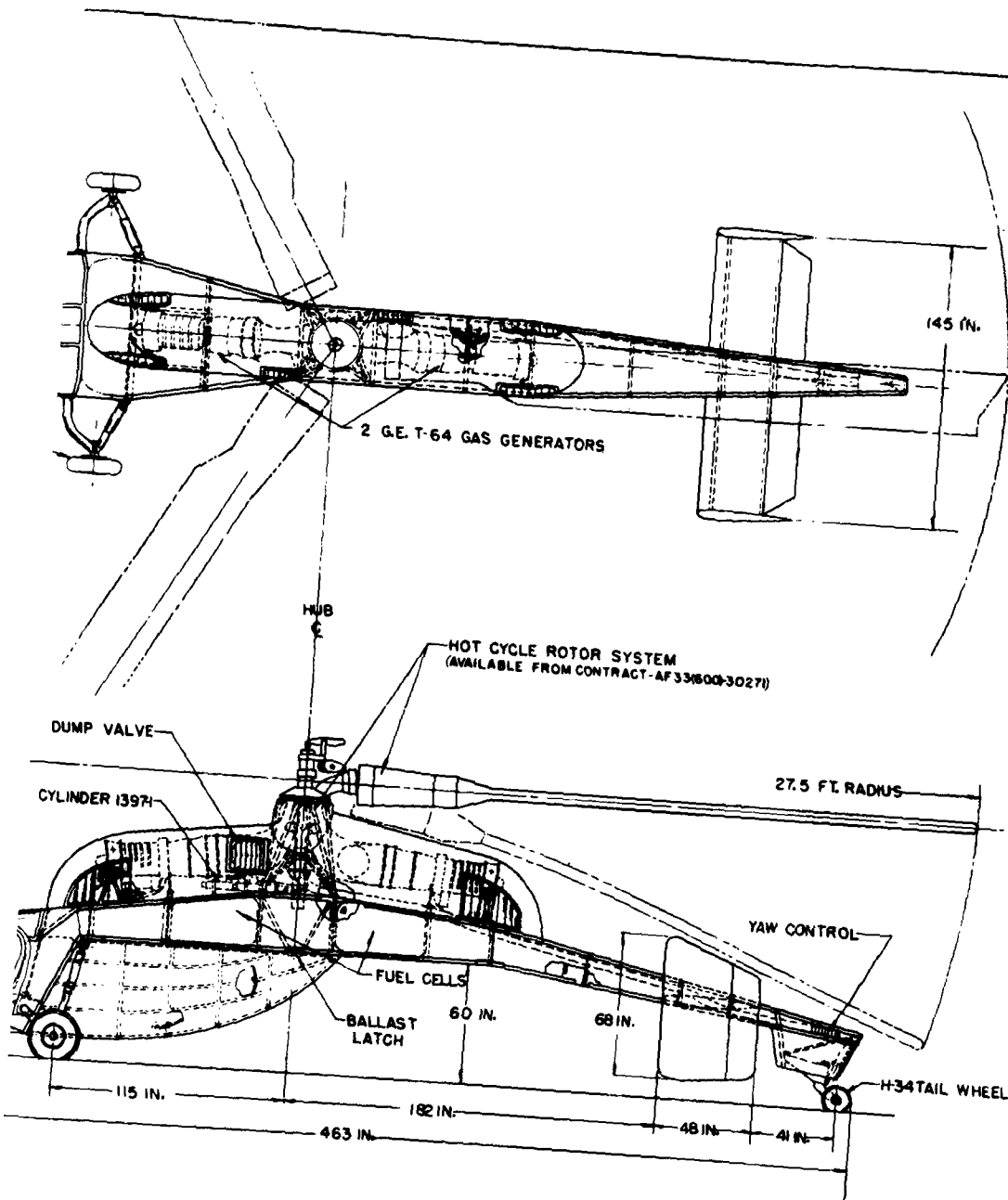
Initial Crane-Type Research Aircraft

c. Cost analysis

d. Single curvature surface for fuselage and tail where possible to reduce cost.

3.3.6   Adaptability to High-μ Flight Investigations

a. Low drag to obtain high speed

b. Maximum jet thrust as an autogyro with hot jet clear of structure

c. Freedom from buffeting and flow separation

d. Good inlet conditions at high speeds.

3.3.7   Adaptability to Heavy-Lift Crane Demonstration

Cargo hook attachment provisions near vertical and horizontal CG location.

3.3.8   Ease of Maintenance

a. Accessibility to engines, rotor, and controls

b. Structural simplicity.

3.3.9   Safety Considerations

a. Pilot vision

b. Fire isolation

a. Crash safety.

3.4   USE OF J85 TYPE DIVERTER VALVES

The General Electric J85 diverter valve shown in Figures 2(a) and 2(b) was investigated shortly after this contract was started. An important characteristic of this valve (which is oversize and heavier than a special T64 diverter valve would be) is that it produces approximately the same low-pressure drop for diverted flow as for straight-through flow. As a result, it can be used in the diverted position to duct gases to the helicopter rotor without serious pressure drops and power losses. In the straight-through flow condition, if both engines are located forward of the rotor and if the gases are directed properly to the rear, as is possible with the valve shown in Figure 2, an

Figure 2a. General Electric J85 Diverter Valve, General Arrangement.
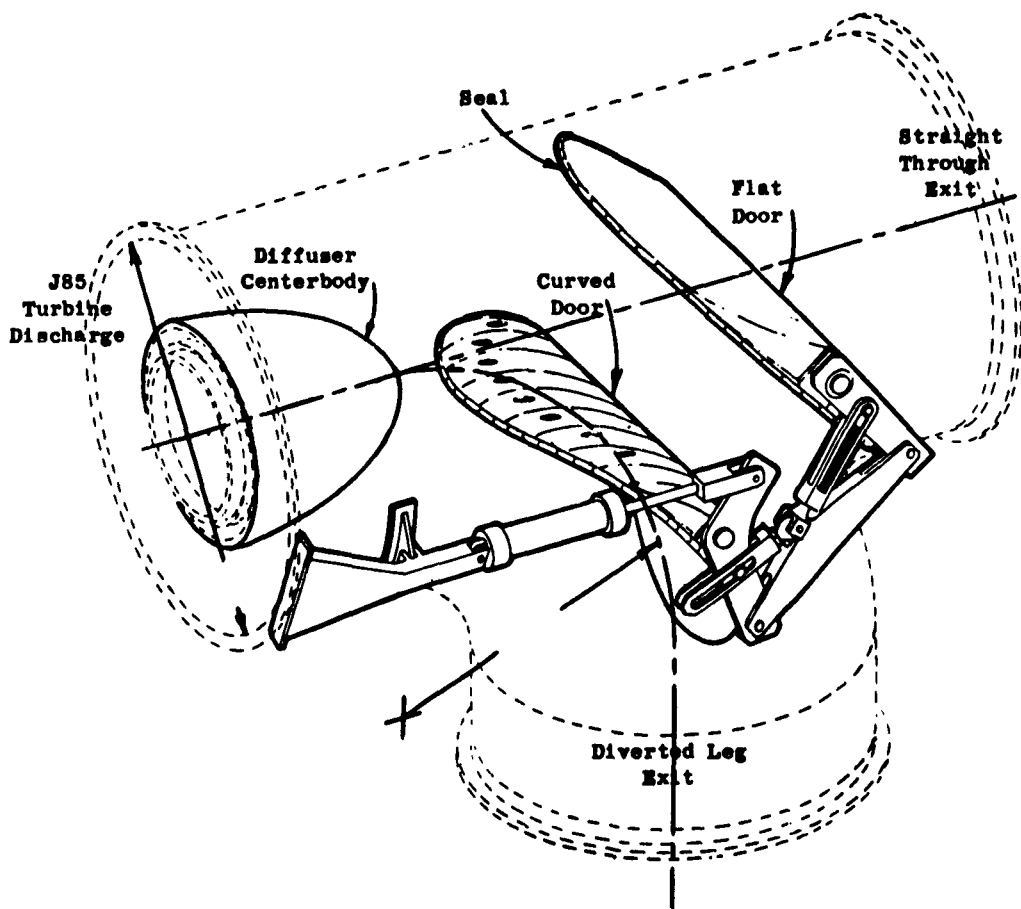
Figure 2b. General Electric J85 Diverter Valve, Interior Arrangement.

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appreciable jet thrust adequate for high-speed autogyro flight would be available. The original diverter valve configuration shown in Figure 1 would have to be a special valve, not now available, which would divert gases to the side and send them to the rotor in the straight-through position. Thus, autogyro flight could not readily be obtained. In addition, the geometry of the special valve of Figure 1 would force location of the center of gravity of the engines further from the rotor center line to get gases to the rotor than would be necessary if J85 valves, in the diverted position, were used. As a result, it would be very difficult to balance a helicopter with two forward-facing engines and special T64 size diverter valves that led the engine gases to the rotor in this straight-through condition. The use of the J85 valves would, on the other hand, permit more freedom in configuration study than could be obtained with a special diverter valve of the Figure 1 configuration.

3.5　　　CONFIGURATION FEATURES INVESTIGATED

　　　　Based on the considerations outlined in the preceding two sections, the configuration features which were investigated before choosing a final configuration were principally the following:

　　　　　　a.　Engine location

　　　　　　b.　Fuselage shape and construction

　　　　　　c.　Cockpit arrangement

　　　　　　d.　Empennage configuration and location

　　　　　　e.　Landing gear configuration and location

　　　　　　f.　Ducting and valves.

Comments on these alternate features are given below.

3.5.1　　Engine Location

3.5.1.1　　Fore and Aft Engines. The initial engine location studied under this contract was the fore and aft arrangement shown on Figure 1. This arrangement was selected on the basis that it would be very easy to obtain proper CG balance. It was originally felt that cruise speeds would be low and that very little ram pressure difference would occur between the fore and the aft engines; therefore, there would be no serious engine performance differences.

　　　　Although the fuselage shown in Figure 1 has a steel tube type of fuselage, this fore and aft engine arrangement could be used

just as readily on a crane with the fuselage construction changed from steel tube to semimonocoque. Such an alternate crane fuselage was considered and is shown on Figure 3(a). This semimonocoque fuselage construction would be similar to that used on the Sikorsky S-64 crane. However, no advantage in engine performance would be gained here by changing the fuselage construction.

More detailed analyses of the engine performance in high-speed forward flight indicated that, if the fore and aft engine locations were used, the rearward-facing engine might experience poor ram recovery because the flow behind the pylon would probably be separated and also cause the flow into the rear engine to be distorted. The forward engine, with its much better ram recovery, might, at the estimated maximum cruise speeds near 200 knots, experience up to 5 percent higher total pressure at the engine inlet with far less distortion across the engine face. The rear engine flow distortion could cause stress problems in the compressor blades of the rear engine, due to fluctuating air loads. The difference in ram pressure could cause differences in control system behavior between the two engines since controls are influenced by inlet pressure.

3.5.1.2   Both Engines Forward. It was therefore seen to be desirable to face both engines forward for performance and stress reasons. A sketch of such an engine arrangement for a flying crane type of fuselage is shown in Figure 3(b). It was found that the aircraft could be balanced properly if both engines were located facing forward through the use of J85 diverter valves.

3.5.1.3   Engine Lateral Location. After the forward location of engines was found to have performance and stress advantages with no sacrifice in ease of balance, attention was given to the lateral location of the forward-facing engines. The following arrangements were considered:

a. Closely spaced engines

b. Partly separated engines

c. Widely separated engines on lateral pylons.

Each of these lateral engine locations was investigated relative to a semimonocoque, low-drag fuselage, which was the final stage of fuselage design development in this study and which is shown in Figure 4.

3.5.1.3.1 Closely Spaced Engines, (Figure 4(a)). In this version, the distance between the two engines is minimum, separated only enough to put the rotor controls between the engine ducting. This close spacing

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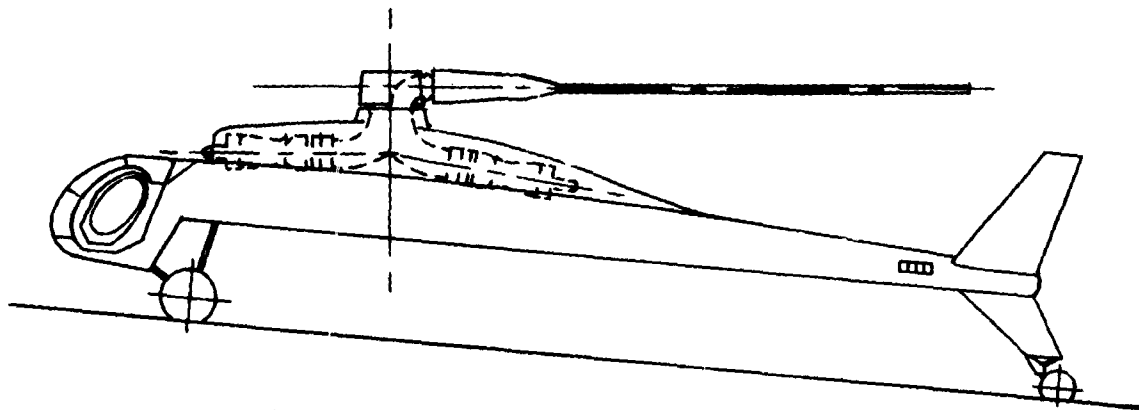
(a) Engines Fore and Aft
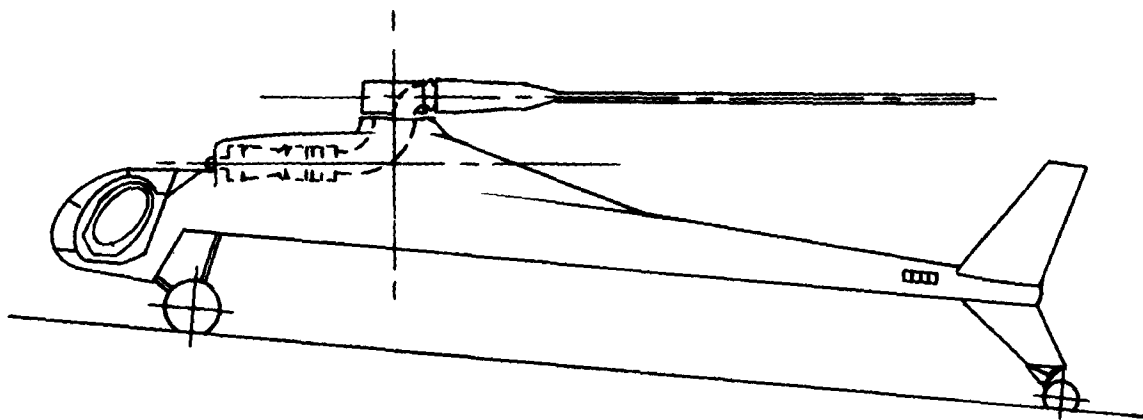
(b) Two Engines Forward

Figure 3. Flying Crane with Semimonocoque Fuselage

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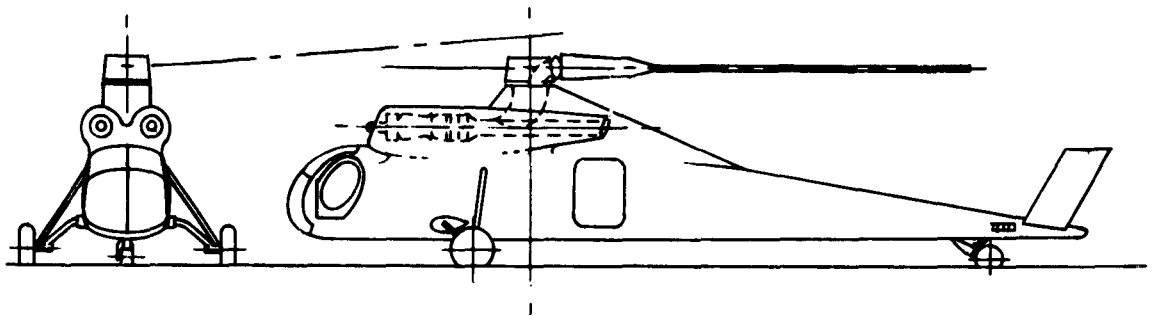
(a) Closely Spaced Engine
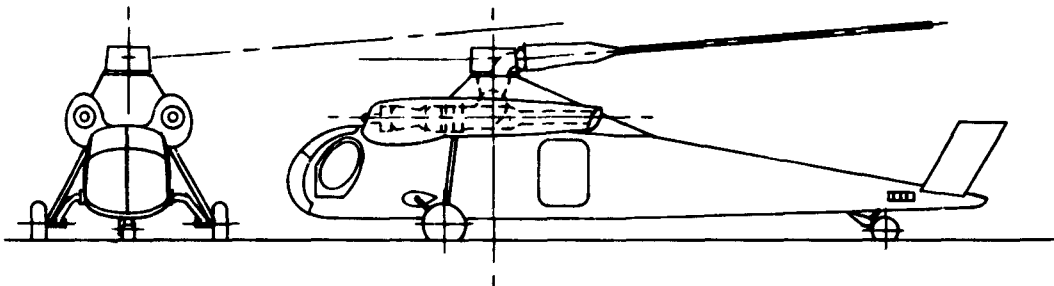
(b) Partly Separated Engines
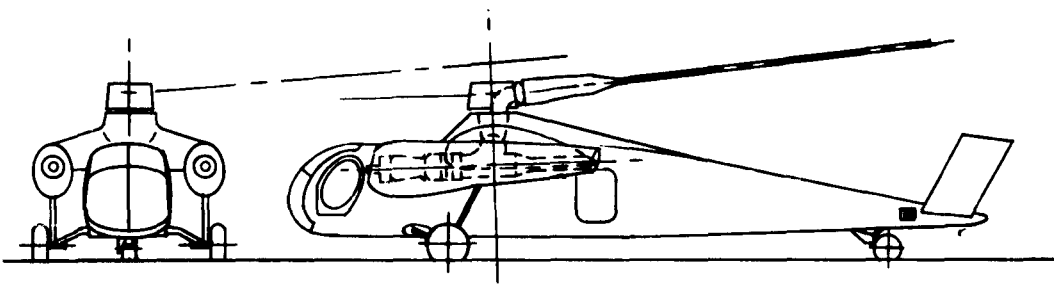
(c) Widely Separated Engines on Lateral Pylons

Figure 4. Low Drag Semimonocoque Fuselage.

precludes the use of the existing "Y" duct to introduce the gas from the two engines to the rotor hub. Instead, a new "Y" duct is required. Also, because of the geometry of the ducting, a special diverter valve would be required in order to duct the flow to the rotor and to permit straight-through flow of gas for evaluation of the autogyro flight regime. Of the three lateral engine locations, this closely spaced version has the lowest frontal area. However, the engine inlets would be located just to the rear and above the pilot's head, so this arrangement would be poor in case of a crash and is unattractive from safety considerations. Furthermore, the limited space between the engines would result in unacceptably poor accessibility to the engine accessories.

3.5.1.3.2 Partly Separated Engines, (Figure 4(b)). This configuration had two substantial changes from the preceding configuration. It used the existing "Y" duct to lead gases into the rotor hub and it also used the existing J85 diverter valve directly coupled to the "Y" duct. Gases were directed to the rotor with the valves in the diverted position and to the rear with the valves in the straight-through position.

Because of the gap between the partly separated engines, the engine diverter valve and "Y" duct combination could be located lower on the fuselage than could the closely spaced arrangement of Section 3.5.1.3.1. In addition, because the J85 valves permit locating the engines closer to the rotor in the fore and aft direction as pointed out in Section 3.4, the cockpit can also be located closer to the rotor. All three fuselages studied in Section 3.5.1.3 had the cockpit located just forward of the engine inlet to give the pilots maximum visibility. With this criterion of cockpit location, the fuselage with the partly separated engines will be shorter forward of the rotor shaft than the fuselage with closely spaced engines. In addition, the length of fuselage aft of the rotor shaft necessary to balance the aircraft (with the same stabilizer area for the closely spaced engine case) will also be less. The net result will be a fuselage that is 315 pounds lighter. Also, because the rotor is lower on the fuselage, the vertical center of gravity is lower, the landing gear tread can be reduced for equivalent overturn angle, and the landing gear weight can be reduced about 40 pounds.

This configuration with partly separated engines will have greatly improved accessibility to the engine accessories compared to the closely spaced engine version, as emphasized in Section 3.3.8, and improved crash safety as emphasized in Section 3.3.9, because the engines will be displaced laterally from being directly behind the pilot's head. However, the parasite area of this fuselage will be increased by approximately 1 square foot due to increased frontal area, wetted surface, and interference between the fuselage and nacelles.

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3.5.1.3.3 <u>Widely Separated Engines on Lateral Pylons (Figure 4(c))</u>.
The last variation of lateral engine placement was location of the engines
at the ends of lateral pylons. The parasite area for this arrangement
was computed to be lower than for the partly separated engine case of
the last section and only slightly worse than the closely spaced engine
version. When the engines are placed in separate nacelles with well-
faired pylons connecting them to the fuselage, the reduction of interfer-
ence drag is enough larger than the increase in wetted area, compared
to the partly separated engines case, that the overall drag is reduced.

Widely separated engines also result in the highest relative
crash and fire safety of the three lateral engine locations. In addition,
the accessibility of engine accessories is far superior to the other two
locations. Even with the addition of the lateral pylons, the widely
spaced engine configuration has an empty weight that is 320 pounds
lighter than that of the closely spaced engine configuration.

3.5.2   <u>Fuselage Construction</u>

There were three variations of fuselage construction con-
sidered in this preliminary design study. They were:

    a. Steel tube crane

    b. Semimonocoque crane

    c. Semimonocoque low-drag.

The steel tube type of fuselage is shown in Figure 1 and represents the
initial approach to design of the research aircraft. A welded fuselage
was proposed because it was thought to be inexpensive. The fuselage
itself was to have been covered with streamlined metal fairings, as
shown in Figure 1, in order to obtain reasonable high speeds as a heli-
copter for test purposes. In addition, a semimonocoque fairing behind
the cockpit was suggested.

Reexamination of this steel-tube-plus-cover-plate fuselage
indicated that a more efficient type of construction for a crane fuselage
would be to use semimonocoque construction (as shown in Figure 3),
because it would be lighter than the first version of Figure 1 with per-
haps the same cost. This type of fuselage construction would be equiv-
alent to that used on the Sikorsky S-60 and S-64 cranes.

The final step in the type of fuselage construction studied
was influenced by the realization that high autogyro cruise speeds could
be obtained with the research aircraft if the overall drag of the helicop-
ter were low enough. A study was made showing that a well-faired,

low-drag fuselage could be designed, with almost all the skin bent in single curvatures, that is relatively easy and inexpensive to form. In fact, as will be pointed out in Section 7.2, Structural Characteristics, the semimonocoque, low-drag fuselage is actually lighter than a crane semimonocoque fuselage would be because the crane fuselage is too shallow for its long length to be efficient. The low-drag fuselage has a better ratio of depth to length, which will result in a fuselage that is lighter and cheaper, besides being vastly cleaner aerodynamically.

3.5.3   Cockpit Arrangement

The research aircraft is to use existing components as much as possible and the preliminary version of the aircraft was shown in Figure 1 with a modified cockpit from a Hughes 269A helicopter. Use of this cockpit (widened to suit the Figure 1 design) would have resulted in appreciable savings compared to designing and building a new one. However, the drag associated with this cockpit would be high enough to prevent reaching the high autogyro cruise speeds that were being sought. Therefore, in conjunction with the use of a semimonocoque, low-drag fuselage, described in Section 3.5.2, a study was made of the use of the cockpit from the Hughes Model 369 helicopter (the Army HO-6). This cockpit is much cleaner aerodynamically than that on the 269A. It was found that it could be mated to a low-drag fuselage with only minor structural changes and no changes in external contour, thus making available a cabin interior and cockpit flight control components at very little cost.

It should be noted that the high autogyro cruise speeds anticipated for the research aircraft will lead to airloads on the plexiglass that are higher than the design loads used in the HO-6 design. However, adequate strength and rigidity for these increased airloads can be obtained for a small increase in weight.

3.5.4   Empennage

The empennage configuration originally considered for the research aircraft consisted of a horizontal tail plus end plates. This tail was to be located at the fuselage station corresponding roughly to 70 percent rotor radius. Thus, the tail would be entirely within the rotor downwash during hovering and would, therefore, experience a substantial download which would result in a smaller net thrust from the rotor. This tail location was chosen originally in spite of download because less abrupt control trim change would be experienced by the pilot than if the tail were located further aft and clear of the rotor downwash.

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Study indicated that if the tail were changed to a vee-type tail instead of a horizontal tail plus end plates, the rotor downwash would gradually, rather than abruptly, cross the tail as the transition was made to forward flight and smooth control motions would result. Further, if the tail were located under the rotor, it would be subjected in hovering to a three-per-rev excitation which might be amplified as a shake of the fuselage. This source of excitation would be removed if the tail were located aft of the rotor.

3.5.5   Landing Gear Configuration and Location

Three landing gear arrangements were considered:

a.   Tail wheel type, with main gear well forward, as the crane of Figure 1

b.   Tail wheel type, with main gear close to the CG, as on the low-drag aircraft of Figure 4

c.   Nose wheel type.

Analysis of the first version, which was predicated on the use of an existing H-34 landing gear, indicated that the extreme forward position of the main landing gear shown in Figure 1 would result in a tail wheel load far higher than that for which it was originally designed. In fact, the load at the rear wheel for this configuration would indicate that the rear wheel should be of the same size and strength and as heavy as one of the main wheels.

By relocating the main landing gear farther aft, closer to the CG, the tail wheel load can be reduced considerably, to the point where the load distribution between the main and rear wheels would be the same as the load distribution on the H-34 landing gear when installed on the H-34 helicopter. Obviously, it was not possible with the fuselage shown in the Figure 1 design to locate the main H-34 landing gear back near the CG because of the absence of supporting structure. If a low-drag fuselage is used, instead of the pod-type crane fuselage, there will be adequate structure near the CG to locate a main landing gear so that the available H-34 landing gear could be used.

A third, nose wheel type of landing gear was considered. Analysis indicates that this type of landing gear will have superior stability for roll-on type landings. However, a nose wheel type is inherently heavier because of the nature of its loading spectrum than a tail wheel type. In addition, the aft structure of an aircraft with nose wheel type landing gear must be designed to take impact loads for nose-high landings. Thus, the nose wheel type gear leads to more

weight forward and aft of the CG, which would result in higher manufacturing cost. Most important, a serious cost penalty would be incurred by the need to develop a new landing gear instead of using available H-34 hardware.

3.5.6 Ducting and Valves

The choice of ducting and valve configuration has been discussed earlier in Sections 3.4, 3.5.1.2 and 3.5.1.3. A brief review will be given here of the significant factors.

3.5.6.1 "Y" Duct.

Since a major objective of this contract is to use existing components, use of the existing stationary "Y" duct which directs gases from the diverter valves into the rotor was investigated. The geometry of this "Y" duct precluded its use in the closely spaced engine configuration of Section 3.5.1.3.1, but it was found that it could be used for partly separated engines, Section 3.5.1.3.2, and for widely separated engines on lateral pylons, Section 3.5.1.3.3. It is advisable to use this duct if possible because it is available (it has been built and tested) and its pressure drop characteristics are known and are reasonable. Use of the "Y" duct does, however, generally increase the frontal area of the aircraft and its parasite area.

3.5.6.2 Diverter Valves.

The choice of diverter valves would be between the existing J85 diverter valve and a new specially designed T64 valve. While the J85 diverter valve is undeniably larger and heavier than a new T64 valve would be, the J85 valve has been built and tested and is available at reasonable cost. It can be used without change to obtain high-speed autogyro flight. A new T64 valve would save some weight and could be used for autogyro flight if so designed. However, it would be quite expensive in time and money to develop a new valve, even allowing for the experience gained by G. E. in developing the J85 diverter valve.

3.5.6.3 Blade Tip Cascade Valves for Single Engine Operation.

It was pointed out in Reference 1 that blade duct valves are needed which effectively reduce the nozzle area at the blade tips to half the total area available if one of the two engines fails, to keep the remaining good engine on its operating line. This reduction of effective nozzle area can be accomplished by valves located at the blade root or at the tip. The root-located valve would be effective because it would shut off completely one of the two ducts located in each blade, each of which has its own proper and fixed nozzle area. The tip-located valve would directly reduce by 50 percent the nozzle area associated with each duct.

Although the root-located valve was originally considered for the Figure 1 version of the research aircraft, because the centrifugal loads on valves at the root would be much less than if they were at the tip, further study of the problem indicated that considerable redesign of blade structure would be required. After further study, a tip-located valve was designed which had essentially mass-balanced doors so that centrifugal force loads were either eliminated or reduced to the values anticipated for the root-located valves. This new approach to the design of valves for one-engine-out operation also results in more rotor power because the effective duct area is doubled, friction losses are correspondingly reduced, and interduct leakage is eliminated. The reduced blade friction pressure drop and elimination of leakage from the active duct result in 35 percent more rotor horsepower than if the root valves were used.

3.6   GENERAL ARRANGEMENT OF SELECTED CONFIGURATION

The general arrangement of the configuration selected for the research aircraft is shown in Figure 5. This configuration was chosen as the optimum compromise of the many alternate design features discussed in Section 3.5. The significant features of this final design are:

a.   Both engines installed forward of the rotor.

b.   Semimonocoque, low-drag fuselage.

c.   Engines widely separated and suspended from lateral pylons extending from the fuselage.

d.   HO-6 type cockpit, strengthened to take the airloads of the expected high cruise speeds.

e.   Vee-type tail located at the aft end of the fuselage clear of the hovering downwash.

f.   Tail-wheel type H-34 landing gear with H-34 load distribution.

3.7   ADVANTAGES OF THE SELECTED CONFIGURATION

The overall advantages that will be obtained with the selected configuration given in Figure 5 and Section 3.6 are summarized below:

a.   Acceptable ducting for gases to the rotor. Good gas flow path for autogyro mode.

b. Longitudinal balance obtained by placement of engine CG close to rotor shaft, permitted by lateral engine location and use of J85 diverter valves.

c. Four hundred-pound download on tail in hovering flight eliminated by aft placement of tail. Fuselage download reduced substantially by smooth contours.

d. Excellent engine inlet conditions for transition and high-speed flight.

e. Smooth transition into and from forward flight attained by gradual immersion of vee-tail into rotor downwash. Fuselage excitation due to fluctuating downwash on tail alleviated by removing empennage from location beneath rotor. Control power enhanced by low CG location.

f. By making the initial design comparatively clean aerodynamically, only a minor cleanup in the areas of landing gear and rotor is required for the high-speed investigations. This is further facilitated by a straight-through tail pipe for jet thrust as an autogyro with jet flow sufficiently separated from fuselage to prevent flow attachment.

g. Fuselage depth selected for maximum structural efficiency and lowest weight.

h. Tail wheel load and fuselage shears minimized by aft placement of main gear and tail gear.

i. Narrower landing gear tread permitted by low silhouette, saving weight and improving ground stability.

j. Cost and schedule advantages result from use of off-the-shelf items, such as; J85 diverter valves, present stationary "Y" duct, hydraulic actuators, H-34 landing gear, and HO-6 type cockpit. Also, minimum tooling can be used for fuselage with single-curvature skin supported by Z-shaped stringers which are, in turn, clipped to simple circular rolled ring frames. Nacelle-type power plants permit low-cost modular construction-fabrication methods.

k. Alternate overload is possible through provisions for cargo hook and hatch.

4'-9"
(57")

J85 DIVERTER VALVE

4'-4"
(52")

10'-4"
(124")

COMPRESSED POSITION
STATIC POSITION
EXTENDED POSITION

3'-1"
(37")

11'-0"
(132")

9.5" AXLE VERTICAL TR

HO-6 COCKPIT

YT 64 GAS GEN

1

Figure 5. General Arrangement - Hot Cycle Research Aircraft.

23

27'-6"
(330")

PRESENT HOT CYCLE ROTOR

COMPRESSED POSITION
STATIC POSITION
EXTENDED POSITION

3'-1"
(37")

24'-5"
(293")

27'-6"
(330")

43'-8"
(524")

XLE VERTICAL TRAVEL

GEN

H-34 MAIN GEAR

H-34 TAIL GEAR

12"AXLE VERTICAL TRAVEL

385-0002

0   50   100   150   200

SCALE- INCHES
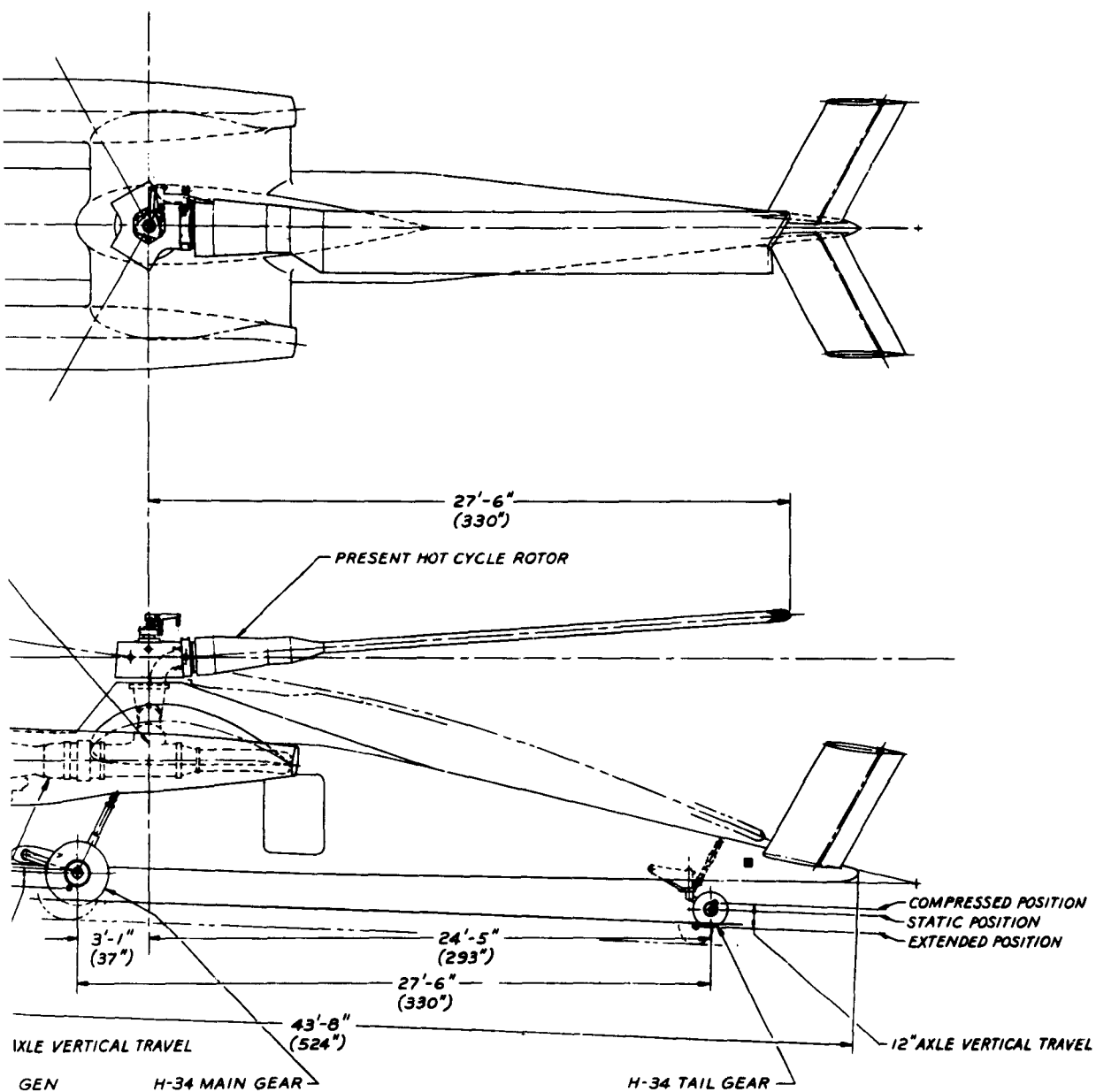
GENERAL
ARRANGEMENT-
HOT CYCLE
RESEARCH AIRCRAFT 385-0002

2

l. Ease of maintenance assured by excellent engine accessibility and all-metal construction.

m. Excellent safety characteristics obtained because of unobstructed pilot vision. Fire and crash safety enhanced by isolated power plants in outboard nacelles.

4. WEIGHT AND BALANCE

Preliminary weight estimates have been made for all major systems of the research aircraft, and the resulting CG has been computed for the empty weight plus the design gross weight and an alternate overload gross weight. This weight and balance information and a discussion of the manner in which the weight estimates were made are presented in the following sections.

4.1   WEIGHT ANALYSIS

The empty weight of the research aircraft has been determined, in general, from four sources: comparative analysis, specific analysis, parametric analysis, and manufacturer's catalogs.

a. By comparative analysis, component weights were obtained that either represent the actual weight of a component as used in a similar application, or represent the component weight after minor design changes required for the new application. In this category, the rotor group weight in the weight statement reflects the actual weights recorded on the whirl test rotor assembly completed in Contract AF 33(600)-30271. In addition, the actual weights of the rotor head control system and of the duct system within the hub envelope for the test rotor are used for the comparable components on the research aircraft. A total of 3330 pounds, or 39 percent of the 8470 pounds empty weight, is actual weight recorded from the whirl test components.

b. By specific analysis, the weights of the Body Group, Tail Group, and Nacelle Group were obtained. A preliminary structural analysis of these components was made and the primary members were sized. The weights were determined by calculation of areas and the application of appropriate factors to account for nonoptimum structure. About 20 percent of the empty weight was accounted for by this method.

c. About 15 percent of the empty weight was determined by parametric analysis. This category includes such systems as hydraulic, electronic, heating and ventilating, etc. The major source of weight information in this category was Reference 2.

d. The remaining 26 percent of empty weight, (approximately 2200 pounds), is catalog weight or actual weight of assemblies used on other service aircraft. It also includes the weight of components "borrowed" from the HO-6 helicopter. This catalog category covers such items as the engines, starters, generators, batteries, instruments,

radio, landing gear (H-34), furnishings, cockpit, and cockpit control system.

The total empty weight of 8470 pounds is somewhat conservative due to the requirement for using the actual whirl test rotor components from Contract AF 33(600)-30271. It was pointed out in Reference 3, Section 3, that approximately 500 pounds could be trimmed from the weights of the rotor group, ducts and seals, and the control system if a redesign of these units were undertaken.

4.2   WEIGHT STATEMENT

SUMMARY WEIGHT STATEMENT
ROTORCRAFT ONLY
(ESTIMATED)

ENGINE	MAIN	AUXILIARY
MANUFACTURED BY	G. E.	
MODEL	T64 gas generator version	
NUMBER	Two (2)	
PROPELLER		
MANUFACTURED BY		
MODEL		
NUMBER		

ROTORCRAFT
SUMMARY WEIGHT STATEMENT
WEIGHT EMPTY
(Pounds)

ROTOR GROUP			2530
BLADE ASSEMBLY	3	1642	
HUB STRUCTURE		634	
GIMBAL		120	
SHAFT		96	
MISCELLANEOUS		38	
FOLDING			
WING GROUP			
CENTER SECTION - BASIC STRUCTURE			
INTERMEDIATE PANEL - BASIC STRUCTURE			
OUTER PANEL - BASIC STRUCTURE (INCL TIPS LBS.)			
SECONDARY STRUCTURE (INCL WING FOLD MECH LBS.)			
AILERONS (INCL BALANCE WT. LBS.)			
FLAPS - TRAILING EDGE			
- LEADING EDGE			
SLATS			
SPOILERS			
TAIL GROUP			120
TAIL ROTOR - BLADES			
- HUB			
STABILIZER		120	
FINS - BASIC STRUCTURE (INCL DORSAL LBS.)			

ROTORCRAFT
SUMMARY WEIGHT STATEMENT
WEIGHT EMPTY
(Pounds)

TAIL GROUP (continued)		
SECONDARY STRUCTURE - STABILIZER AND FINS		
ELEVATOR (INCL BALANCE WEIGHT LBS.)		
RUDDER (INCL BALANCE WEIGHT LBS.)		
BODY GROUP		1140
FUSELAGE OR HULL - BASIC STRUCTURE	1020	
BOOMS - BASIC STRUCTURE	120	
SECONDARY STRUCTURE - FUSELAGE OR HULL		
- BOOMS		
- DOORS, PANELS AND MISC.		

ALIGHTING GEAR GROUP - LAND-TYPE					450
LOCATION (H-34A LANDING GEAR)	*ROLLING ASSEMBLY	STRUCT.	CONTROLS		
MAIN GEAR 11.00 - 12	162	211	17	390	
TAIL WHEEL 6.00 - 6	11	45	4	60	

ALIGHTING GEAR GROUP - WATER-TYPE					
LOCATION	FLOATS	STRUTS	CONTROLS		

* WHEELS, BRAKES, TIRES, TUBES AND AIR

ROTORCRAFT
SUMMARY WEIGHT STATEMENT
WEIGHT EMPTY
(Pounds)

FLIGHT CONTROLS GROUP						860
COCKPIT CONTROLS					30	
AUTOMATIC PILOT					--	
SYSTEM CONTROLS - ROTOR HEAD					550	
- LINKAGE					79	
- HYDRAULIC					96	
- YAW CONTROLS - JET					105	
ENGINE SECTION OR NACELLE GROUP						420
ENGINE MOUNTS					30	
COWLING, STRUCTURE AND FIREWALL					220	
NACELLE SUPPORT PYLONS					170	
DOORS, PANELS AND MISC.						
PROPULSION GROUP						2160
	AUXILIARY			MAIN		
ENGINE INSTALLATION						
ENGINE				1160		
TIP BURNERS						
LOAD COMPRESSOR						
REDUCTION GEAR BOX						
ACCESSORY GEAR BOXES AND DRIVES				20		
SUPERCHARGER FOR TURBO TYPES						
AIR INDUCTION SYSTEM				10		
EXHAUST SYSTEM				40		

ROTORCRAFT
SUMMARY WEIGHT STATEMENT
WEIGHT EMPTY
(Pounds)

PROPULSION GROUP (continued)					
	AUXILIARY		MAIN		
COOLING SYSTEM					
LUBRICATING SYSTEM				70	
TANKS			4		
COOLING INSTALLATION			54		
PLUMBING, ETC.			12		
FUEL SYSTEM				280	
TANKS-PROTECTED			--		
-UNPROTECTED			200		
PLUMBING, ETC.			80		
WATER INJECTION SYSTEM					
ENGINE CONTROLS				50	
STARTING SYSTEM				70	
PROPELLER INSTALLATION					
DRIVE SYSTEM					
GEAR BOXES					
CLUTCH AND MISC					
TRANSMISSION DRIVE					
ROTOR SHAFT					
JET DRIVE				460	
DUMP VALVES (2)			150		
CONNECTORS			20		
DUCTS AND SEALS IN HUB			290		
AUXILIARY POWER PLANT GROUP					

ROTORCRAFT
SUMMARY WEIGHT STATEMENT
WEIGHT EMPTY
(Pounds)

INSTRUMENT AND NAVIGATIONAL EQUIPMENT GROUP		50
INSTRUMENTS	49	
NAVIGATIONAL EQUIPMENT	1	
HYDRAULIC AND PNEUMATIC GROUP		210
HYDRAULIC		
PNEUMATIC		
ELECTRICAL GROUP		330
A C SYSTEM		
D C SYSTEM		
ELECTRONICS GROUP		10
EQUIPMENT		
INSTALLATION		
ARMAMENT GROUP (INCL GUNFIRE PROTECTION PROTECTION LBS.)		--
FURNISHINGS AND EQUIPMENT GROUP		80
ACCOMMODATIONS FOR PERSONNEL	22	
MISCELLANEOUS EQUIPMENT (INCL LB BALLAST)		
FURNISHINGS	8	
EMERGENCY EQUIPMENT	50	
AIR CONDITIONING AND ANTI-ICING EQUIPMENT		60
AIR CONDITIONING	50	
ANTI-ICING	10	

ROTORCRAFT
SUMMARY WEIGHT STATEMENT
WEIGHT EMPTY
(Pounds)

PHOTOGRAPHIC GROUP		
EQUIPMENT		
INSTALLATION		
AUXILIARY GEAR GROUP		50
AIRCRAFT HANDLING		
LOAD HANDLING	50	
A TO GEAR		
MANUFACTURING VARIATION		
TOTAL - WEIGHT EMPTY		8470

ROTORCRAFT
SUMMARY WEIGHT STATEMENT
WEIGHT EMPTY
(Pounds)

LOAD CONDITION			DESIGN	ALTERNATE OVERLOAD	
CREW - NO.			400	400	
PASSENGERS - NO.					
FUEL	TYPE	GALS.			
UNUSABLE					
INTERNAL		500	3250		
		200		1300	
EXTERNAL					
BOMB BAY					
OIL					
UNUSABLE			5	5	
ENGINE			30	30	
BAGGAGE					
CARGO			3145	15045	
ARMAMENT					
GUNS - LOCATION FIX/FLEX QUANTITY CALIBER					
AMMUNITION					
·INSTALLATIONS - BOMB, TORPEDO, ROCKET, ETC.					
* BOMB OR TORPEDO RACKS					

* IF NOT SPECIFIED AS WEIGHT EMPTY

ROTORCRAFT
SUMMARY WEIGHT STATEMENT
WEIGHT EMPTY
(Pounds)

LOAD CONDITION (continued)	DESIGN	ALTERNATE OVERLOAD	
EQUIPMENT			
PYROTECHNICS			
PHOTOGRAPHIC			
CARGO SLING		250	
*OXYGEN			
MISCELLANEOUS			
USEFUL LOAD	6830	17030	
WEIGHT EMPTY	8470	8470	
GROSS WEIGHTS	15300	25500	

*IF NOT SPECIFIED AS WEIGHT EMPTY

4.3   <u>WEIGHT AND BALANCE STATEMENT</u>

SUMMARY

WEIGHT AND BALANCE

	Weight Lb.	C. G.	
		Fwd. ₵ Rotor Inches	Below Rotor Plane Inches
WEIGHT EMPTY	8470	2.8	41.0
USEFUL LOAD	6830		
DESIGN GROSS WEIGHT	15300	3.3	55.2
USEFUL LOAD - Alternate overload	17030		
GROSS WEIGHT - Alternate overload	25500	2.3	60.6

REFERENCE DATA:

    1.   Ref. datum for horizontal C. G. is ₵ rotor.

    2.   Ref. datum for vertical C. G. in rotor plane (0° cone).

WEIGHT EMPTY SUMMARY

Item and Remarks	Weight 1	Weight 2	Horiz. Dist. ℄ Rotor	H Moment 1	H Moment 2	Vert. Dist. ℄ Rotor	V Moment 1	V Moment 2
ROTOR GROUP								
Blades - 27.5' Rad - 31.5" Chd. (3)	2530	1642						
Gimbal Assembly		120						
Hub Structure		634						
Shaft		96						
Miscellaneous		38						
TAIL GROUP	120		348	41760		- 78	- 9360	
BODY GROUP	1140		(17.)	19385		(- 75.77)	86375	
Cockpit Section		100	- 118		- 11800	- 80		- 8000
Main Fus. Section		840	5		4200	- 77		- 64680
Boom Section		120	233		26760	- 92		- 11040
Pylon or Rotor Mount		40				- 27		- 1080
Pylon Fairing		15				- 30		- 450
Fuselage Carry-Thru		25	15		225	- 45		- 1125
ALIGHTING GEAR GROUP (H-34A)	450		(3.98)	1789		(-103.8)	-46720	
Rolling Gear - Main		162	- 37		- 5994	-112		-18144
Chassis - Main		167	- 36		- 6012	- 98		-16366
Brake Oper. Mech.		17	- 50		- 850	-104		- 1768
Tail Wheel		11	293		3223	-129		- 1419
Chassis -. Tail Wheel		42	278		11676	-114		- 4788
Locking Mech. - T.W.		3	282		846	-117		- 351
Install. - Main Gear		44	- 50		- 2200	- 78		- 3432
- Tail Gear		4	275		1100	-113		- 452

WEIGHT EMPTY SUMMARY

Item and Remarks	Weight		Horiz. Dist. ₵ Rotor	H Moment		Vert. Dist. ₵ Rotor	V Moment	
1 2	1	2		2	1		2	1
FLIGHT CONTROLS GROUP	860		(14.8)		12770	(- 33.7)		-28960
Cockpit (dual)		30	-120	- 3600		- 90	- 2700	
Intermediate Linkage		100	- 55	- 5500		- 45	- 4500	
Rotor Head Controls		550				- 20	-11000	
Hyd. Cyl. and Servos (3)		75	- 16	- 1200		- 40	- 3000	
Jet Yaw Control - Pipe		75	150	11250		- 62	- 4650	
- Tail Unit		20	510	10200		-103	- 2060	
- Controls		10	162	1620		-105	- 1050	
NACELLE	420		(- 14.6)		- 6150	(- 45.4)		-19060
Engine Mounts		30	- 60	- 1800		- 52	- 1560	
Firewall		20	- 40	- 800		- 52	- 1040	
Cowling and Struc.		200	- 22	- 4400		- 52	-10000	
Nacelle Support Stubs		170	5	850		- 38	- 6460	
PROPULSION GROUP	2160		(- 34.0)		-73430	(- 50.2)		-108640
Engines - 2 - T-64		1160	- 59	-68440		- 53	-61480	
Air Inlets		10	- 90	- 900		- 52	- 520	
Exhaust Pipes		40	47	1880		- 52	2080	
Accessory Gear Box & Drive		20	15	300		- 35	- 700	
Lubrication System		70	- 20	- 1400		- 32	- 2240	
Tanks (2)		(4)						
Coolers and Supports		(30)						
Ducts and Blowers		(24)						
Distrib. System		(12)						
FUEL SYSTEM		280	6	1680		- 76	-21280	
Tanks (500 Gal.) & Supports		(200)						
Pumps		(10)						
Distribution System		(70)						

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WEIGHT EMPTY SUMMARY

Item and Remarks	Weight 1	Weight 2	Horiz. Dist. ℄ Rotor	H Moment 2	H Moment 1	Vert. Dist. ℄ Rotor	V Moment 2	V Moment 1
STARTING SYSTEM								
Starters (2)		70	- 50	- 3500		- 60	- 4200	
Controls		(60)						
		(10)						
POWER PLANT CONTROLS		50	- 40	- 2000		- 60	- 3000	
DUCTING - ROTOR DRIVE		460						
Dump Valves (2)		150	- 7	- 1050		- 49	- 7350	
Connectors		(20)	0			- 43	- 860	
Ducts and Seals in Hub		(290)				- 17	- 4930	
INSTRUMENT & NAV. GROUP	50				- 5250			- 3290
Instruments		32	-140	- 4480		- 72	- 2304	
Transmitters		12	- 30	- 360		- 50	- 600	
Installation		5	- 60	- 300		- 60	- 300	
Map Case		1	-110	- 110		- 86	- 86	
ELECTRICAL EQUIPMENT	330				- 9340			-25410
Generators (2)		50	- 80	- 4000		- 65	- 3250	
Batteries (2)		150				-100	-15000	
Battery Support		3				-100	- 300	
Ext. Power Receptacle		2				-100	- 200	
Gen. Control Box		14	- 50	- 700		- 65	- 910	
Circuitry		101	- 40	- 4040		- 50	- 5050	
Lights		10	- 60	- 600		- 70	- 700	
ELECTRONIC GROUP EQUIP.	10		-100		- 1000	- 85		- 850
HYDRAULIC SYSTEM		210	+ 20		+ 4200	- 35		- 7350

WEIGHT EMPTY SUMMARY

Item and Remarks	Weight 1	Weight 2	Horiz. Dist. ℄ Rotor	H Moment 2	H Moment 1	Vert. Dist. ℄ Rotor	V Moment 2	V Moment 1
FURNISHINGS & EQUIPMENT	80				- 3400			- 2268
Seat Cushions		10	-106	-1060		-74	- 740	
Belts and Harness		12	-105	-1260		-74	- 888	
Instrument Board		8	-135	-1080		-80	- 640	
Fire Ext.		50						
AIR CONDITIONING EQUIP.	60				- 5200			- 4400
Heat and Vent		50	- 80	-4000		-80	-4000	
Windshield Defrost		10	-120	-1200		-40	- 400	
AUXILIARY GEAR	50					-85		- 4250
TOTAL - WEIGHT EMPTY	8470		(- 2.8)		-23866	(-41.0)		-346893

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DESIGN GROSS WEIGHT

Item and Remarks		Weight		Horiz. Dist. ℄ Rotor	H Moment		Vert. Dist. ℄ Rotor	V Moment	
1	2	1	2		2	1		2	1
WEIGHT EMPTY		8470		(- 2.8)		-23866	(-41.0)		-346893
Pilot			200	-110	- 22000		-70	- 14000	
Co-Pilot			200	-110	- 22000		-70	- 14000	
Oil			35	- 20	- 700		-30	- 1050	
Fuel - Fwd. Tank 260 Gal.			1690	- 64	-108160		-73	-123370	
Fuel - Aft Tank 240 Gal.			1560	+ 81	126360		-80	-124800	
Payload			3145				-70	-220150	
TOTAL - USEFUL LOAD		6830				-26500			-497370
GROSS WEIGHT		15300		(- 3.3)		-50366	(-55.2)		-844263

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ALTERNATE OVERLOAD GROSS WEIGHT

Item and Remarks	Weight		Horiz. Dist. ℄ Rotor	H Moment		Vert. Dist. ℄ Rotor	V Moment	
	1	2		2	1		2	1
PILOT		200	-110	-22000		-70	14000	
CO-PILOT		200	-110	-22000		-70	14000	
OIL		35	- 20	- 700		-30	1050	
FUEL - Fwd. Tank 100 Gal.		650	- 64	-41600		-73	47450	
FUEL - Aft. Tank 100 Gal.		650	+ 81	52650		-80	52000	
CARGO SLING		250				-70	17500	
PAYLOAD		15045				-70	-1053150	
USEFUL LOAD	17030				-33650			-1199150
WEIGHT EMPTY	8470				-23866	(-41.0)		- 346893
GROSS WEIGHT	25500		(- 2.3)		-57516	(-60.6)		-1546043

5. PERFORMANCE

In this section are presented the results of computations of the hovering and level flight performance of the research aircraft in the helicopter mode and level flight performance in the autogyro mode.

5.1   ESTIMATION OF PARASITE AREA

Based on the configuration and size of the research aircraft as shown in Figure 5, the total equivalent parasite drag area of the aircraft was estimated to be 22.0 square feet. A breakdown of the equivalent parasite drag area of the individual components is presented in Table I. Drag coefficients and corresponding drag areas used are also presented in Table 1. Reference 4 is used to determine the parasite drag of the rotor head. References 5 and 6 are used to estimate the parasite area of the remaining components.

5.2   PRESSURE DROPS IN DUCTING

Computation of helicopter rotor power available and autogyro thrust available depend on pressure drops in the ducting between the gas generators and the nozzles involved. A special kind of analysis is necessary to compute the pressure drop in the rotating helicopter blades, and that component is discussed in the next section. Information is presented here on the pressure drop between the gas generator and the blade root, with the diverter valve in the diverted position. The pressure drop is also given between the gas generator and the autogyro nozzles, with the diverter valve in the straight-through position.

Figure 6 presents a sketch of the two gas paths involved, including a designation of stations, for convenience. It also includes the results of the pressure drop calculations for the two alternate paths. An evaluation is also included of the influence of fairing into the diverter valve inlet bullet that is part of the standard J85 diverter valve, but which could be omitted if desired. The pressure drop in the diverter valve for the straight-through or diverted position is taken from Reference 7. The pressure drop in the various lengths of straight pipe was based on classical pipe flow analysis. Allowance was made for the effective bellmouth inlet of the straight (autogyro mode) ducts. Pressure drop in the inverted "Y" duct at the rotor head is based on the earlier Hughes component tests of Reference 8.

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TABLE 1 PARASITE DRAG BREAKDOWN			
Components	Applicable Areas as Indicated ft^2	Drag Coefficient	Equivalent Drag Area ft^2
Rotor Head			
Hub	6.25 (max. frontal area)	0.75	4.70
Blade shanks	12.2 (max. frontal area)	0.25	3.05 (1.55)*
Fuselage	23.0 (max. frontal area)	0.085	1.96
Landing Gears			
Wheels	5.00 (max. frontal area)	0.30	1.50
Tail wheel	2.00 (max. frontal area)	0.50	1.00 } (3.70)*
Struts	4.30 (max. frontal area)	1.20	5.20
Empennage	54.00 (surface area)	0.02	1.08
Pylon	14.00 (max. frontal area)	0.06	0.84
Nacelles	13.20 (max. frontal area)	0.05	0.66
			19.99 (14.49)*
Interference, Roughness and Misc. 10%			2.00 (1.45)*
Total equivalent parasite area			21.99 (15.99)*
*Fairings added.			

Sta.(5.23)
Sta.(5.22)
Sta.(5.3)
Sta.(5.31)
12"
32"
Sta.(5.2)
16.5"
17"
16.5"
17"
34.8"
7"
Sta.(5.0)
Sta.(5.1)
Sta.(5.11)
Sta.(5.4)
Sta.(5.41)

Station Number		Station Description	Per Cent, Total Pressure loss between Stations, %	
Straight Thru Operation	Diverted operation		Without The Valve Bullet	With The Valve B
5.0-5.1 or 5.11 —		Turbine Exit-Valve inlet or diffu. Exit	.764	.691
5.1 or 5.11 - 5.4 —		Valve inlet or diffu. Exit - Valve Exit	.195	.187
5.4-5.41	—	Valve Exit - Tailpipe inlet	.298	.295
5.41 - 6	—	Tailpipe inlet - Nozzle inlet	.464	.459
6 - 7	—	Nozzle inlet - Nozzle Exit	—	—
	5.0 -5.1	Turbine Exit - Valve inlet	.764	—
	5.1 - 5.2	Valve inlet - Valve Exit (Diverted)	.262	—
	5.2 - 5.21	Valve Exit - Reducer Exit	.045	—
	5.21 - 5.22	Reducer Exit - Elbow inlet	.091	—
	5.22 - 5.23	Elbow inlet - Elbow Exit	.295	—
	5.23 - 5.3	Elbow Exit - Tri-duct Exit	.297	—
Total			1.731	1.632
	Total		1.764	—

Figure 6.   Duct Station Description and Pressure Losses.

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Sta. (5.3)

12"

32"

16.5"

11"

7"

60"

5"

Sta.(5.4)   Sta.(5.41)   Sta.(6)   Sta.(7)

on	Per Cent Total Pressure loss between stations, % $\frac{\Delta P_t}{P_t}$		Remarks
	Without The Valve Bullet	With The Valve Bullet	
...ordiffu.Exit	.764	.691	
...Valve Exit	.195	.187	
...inlet	.298	.295	
...le inlet	.464	.459	
...le Exit	—	—	
...inlet	.764	—	
...t (Diverted)	.262	—	
...Exit	.045	—	
...nlet	.091	—	
...it	.295	—	
...Exit	.297	—	
	1.731	1.632	
	1.764	—	

FIGURE 6 - DUCT STATION DESCRIPTION
AND PRESSURE LOSSES

NOTE:
All losses are based on the following:
Flow Rate, $\dot{w} = 25$ #/sec.
Total temperature, $T_{t_6} = 1577°R$
Total Pressure, $P_{t_6} = 41.5$ psia.
Mach No. $M_6 = .33$
Gas Const., $R = 53.35$ #/R° $\delta = 1.33$

As seen in Figure 6, for diverted flow to the rotor,

$$\frac{\Delta P_{T_{5-5.3}}}{P_{T_5}}$$

is 1.76 percent. For straight through (undiverted) flow to the autogyro nozzles,

$$\frac{\Delta P_{T_{5-7}}}{P_{T_5}}$$

is 1.73 percent. Since the performance computations of rotor power available have been based on a pressure loss to the rotor blade root of 4 percent, it is seen that the prediction of helicopter power should be conservative by a few percent. The autogyro thrust available, discussed below, was based on a duct pressure loss of 3 percent, which is also somewhat conservative.

5.3   HELICOPTER POWER AVAILABLE

The procedure for computation of helicopter rotor power available in hovering and forward flight is described in Reference 9. It is pointed out that the problem is essentially that of computing the pressure change between the blade root and blade tip. This is a classical thermodynamic problem of flow with friction, heat transfer, and external work. The validity of the performance calculation method was established by the correlation between predicted and measured rotor performance reported in Reference 10, which summarizes the hot cycle rotor whirl tests of Contract AF 33(600)-30271.

As discussed in Reference 9, rotor power is a function of engine gas conditions, pressure drop from engine to blade root (Section 5.2), duct area, duct friction coefficient, blade tip speed, and nozzle velocity coefficient. Based on G. E.-supplied gas conditions (References 11 and 12) for the current T64 engine, which has a 2850 horsepower, 10-minute take-off rating (Reference 13), computations were made of the available rotor power and resulting rotor thrust for take-off at 6000 feet, 95 degrees Fahrenheit. In addition, rotor power available and specific fuel consumption for full and part throttle at sea level standard were also computed.

For the computations of rotor power available, a duct friction coefficient of $f = 0.003$ and a nozzle velocity coefficient of $C_{V_e} = 0.98$ were used, based on analysis of the data taken during the whirl test reported in Reference 10. Using the gas conditions for the take-off rating of the engine as the starting point, a duct inlet Mach number of 0.361 was computed. The resulting pressure ratio across the blade, $(\frac{\text{tip pressure}}{\text{root pressure}})$, was 1.024.

After making allowance for the pressure drop in the ducting from the engine to the blade root and the pressure change in the blade, the power available at 6000 feet, 95 degrees Fahrenheit was found to be 2110 horsepower, which will allow a helicopter hovering gross weight of 18,215 pounds. This value of rotor thrust is substantially higher than the minimum gross weight of 15,000 pounds stipulated in the statement of work of this contract.

The rotor power available at sea level is shown on Figure 7 for the take-off, military, and normal continuous power conditions. Rotor power for take-off at sea level static is about 3050 horsepower. This value of power is shown in Figure 7 as a constant value versus forward speed. This is approximately true at speeds below 200 knots because the extra power that is developed due to ram pressure rise at the engine is just about compensated for by the power required to overcome the ram drag of accelerating the air to helicopter forward speed.

5.4   HELICOPTER LEVEL FLIGHT PERFORMANCE

The power required for helicopter level flight was computed using the parasite area established in Section 5.1 and standard NACA helicopter performance calculation procedures summarized in Reference 14. Figure 7 shows power required versus speed at sea level for 10,000 pounds and 15,300 pounds gross weight. For test purposes, it has been established that a gross weight of 10,000 pounds will be sufficient to allow reasonable instrumentation and fuel load for short missions. Using the take-off power rating, which is good for 10 minutes and is long enough to get stabilized speed data, a maximum helicopter speed at sea level at 10,000 pounds gross weight of 156 knots is predicted.

5.5   AUTOGYRO THRUST AVAILABLE

The autogyro forward flight thrust available was computed using standard thermodynamic procedures, based on gas conditions compatible with the T64 gas generator defined by References 11 and 12, and the tail pipe configuration of Figure 6. For conservatism, a tail pipe pressure loss of 3 percent was assumed, which is appreciably greater than the computed value of 1.73 percent given for the autogyro

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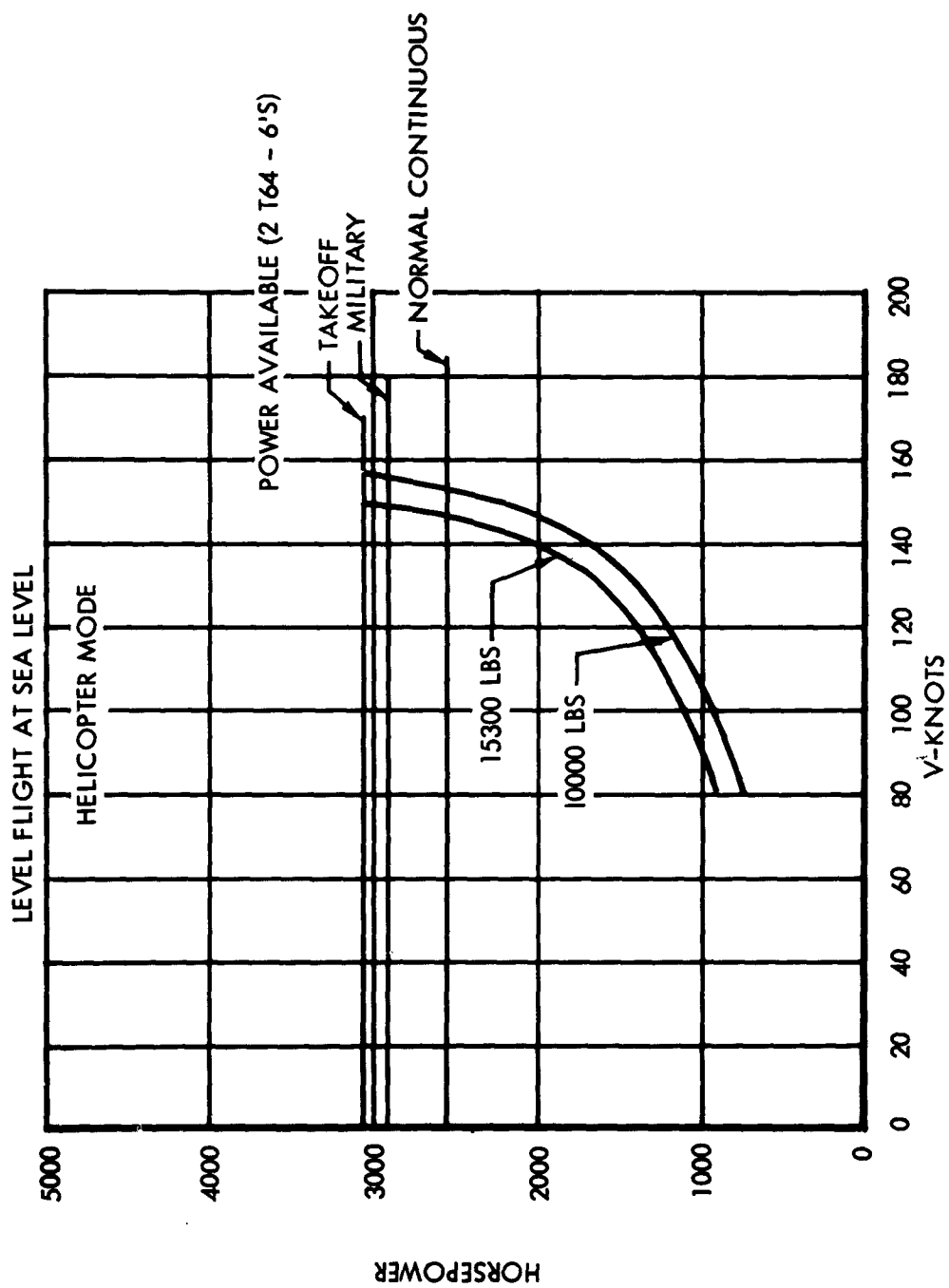
Figure 7. Power Vs. Velocity.

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ducting. The results of these calculations of thrust available are shown on Figure 8 for the take-off, military, and normal continuous power setting. It is seen that the static thrust of the two T64's is about 3500 pounds, which falls to about 3150 pounds at 200 knots.

5.6 AUTOGYRO LEVEL FLIGHT PERFORMANCE

The thrust required for autogyro level flight was computed using the parasite area of Section 5.1, and the procedures of Reference 14 for an unpowered rotor were used to obtain the overall aircraft (L/D) ratio. The autogyro calculations were done only for 10,000 pounds gross weight because it is expected that the maximum speed performance will be sought only at the low weights which permit higher speeds.

A basic lift/drag ratio and resulting thrust required were computed for a tip speed of 700 feet per second, using the parasite area f = 22 obtained in Section 5.1. The 700 feet per second value of tip speed corresponds to the optimum helicopter tip speed based on reasonable helicopter tip stall speeds. However, since the propulsion requirement is divorced from the lifting requirement as far as the rotor is concerned, the rotors on autogyros can generally operate at lower tip speeds than comparable helicopters, resulting in less equivalent overall drag. Lower autogyro tip speeds of 600 feet per second and 500 feet per second were therefore investigated. With the parasite area of f = 22 square feet of Section 5.1, it was found (and shown in Figure 8) that an increase of maximum speed at take-off thrust of about 20 knots (152 to 172 knots) can be achieved if tip speed is reduced to 600 feet per second. If tip speed were reduced to 500 feet per second, only about 4 knots further speed increase would be possible. Since the possibility existed of encountering rotor dynamic problems at this lower tip speed, the thrust required at 500 feet per second was not shown on Figure 8.

Because the autogyro mode does offer such attractive performance in the basic configuration shown in Figure 5, a brief investigation was made of the potential performance gains to be obtained by drag reduction. Study of the Figure 5 general arrangement and the drag breakdown of Section 5.1 led to the conclusion that it should be possible to cut the parasite area from 22 square feet to 16 square feet if simple fairings were placed on the blade shanks and landing gear. With these fairings, the research aircraft is estimated to have a maximum level flight speed of 197 knots, as shown in Figure 8 for take-off thrust.
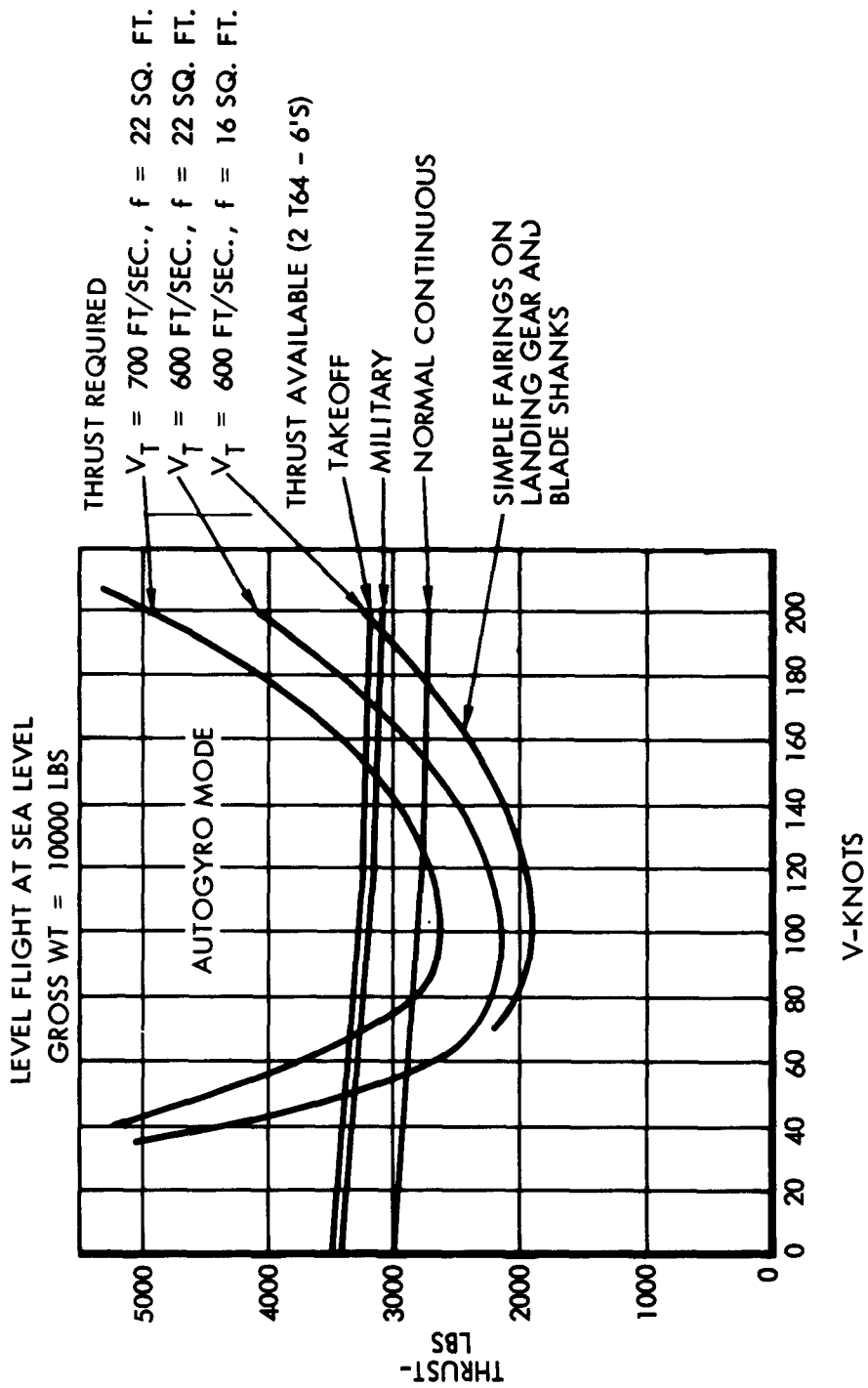
Figure 8. Thrust Vs. Velocity.

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6. STABILITY AND CONTROL

The requirements of SPEC MIL-H-8501A (Reference 15) were used as the stability and control objectives. Analysis of the over-all stability and control of the research aircraft with the configuration of Figure 5 indicates that the aircraft will be controllable in the hands of pilots of reasonable skill. However, because existing components are used wherever possible, the aircraft will not have stability char-acteristics in strict compliance with the requirements of MIL-H-8501A. Future redesign in the area of the hub and control system, to permit use of moderate hub elastic restraint, would permit substantial im-provement in stability and control.

6.1   HOVERING FLIGHT

The expected handling characteristics of the research air-craft in hovering flight using available components were calculated (open symbols in Figure 9). It was found that, with minor adjustments of the control gearing between stick motion and blade pitch change com-pared to those used in the whirl test, the research aircraft will have handling characteristics that make the aircraft safe and reasonably easy to fly (shown as solid symbols in Figure 9). The following control power changes were recommended, primarily to improve the handling quali-ties and to approach the flying qualities criteria of References 15 and 16.

a. Increase the longitudinal control power by reducing the longitudinal cyclic stick travel from the present 12 inches to 9 inches but maintaining the present cyclic pitch travel of ± 10 degrees.

b. Modify the lateral control power as shown in Figure 10 by providing a nonlinear stick-swashplate gearing but maintaining the lateral cyclic pitch travel of ± 7 degrees.

Acceptable boundaries as a function of aircraft damping, control power, and inertia are presented in Figure 9.

6.1.1   Handling Characteristics in Pitch

The open symbols on the pitch curve of Figure 9 represent the handling characteristics of the present hot cycle aircraft in hover. The solid symbols represent the handling characteristics with the in-crease in the longitudinal control power recommended in a. of Section 6.1. It can be seen that the solid points, though improved, are still below the requirements of SPEC MIL-H-8501A due to the low damping characteristics of the aircraft. However, it is felt that the aircraft can

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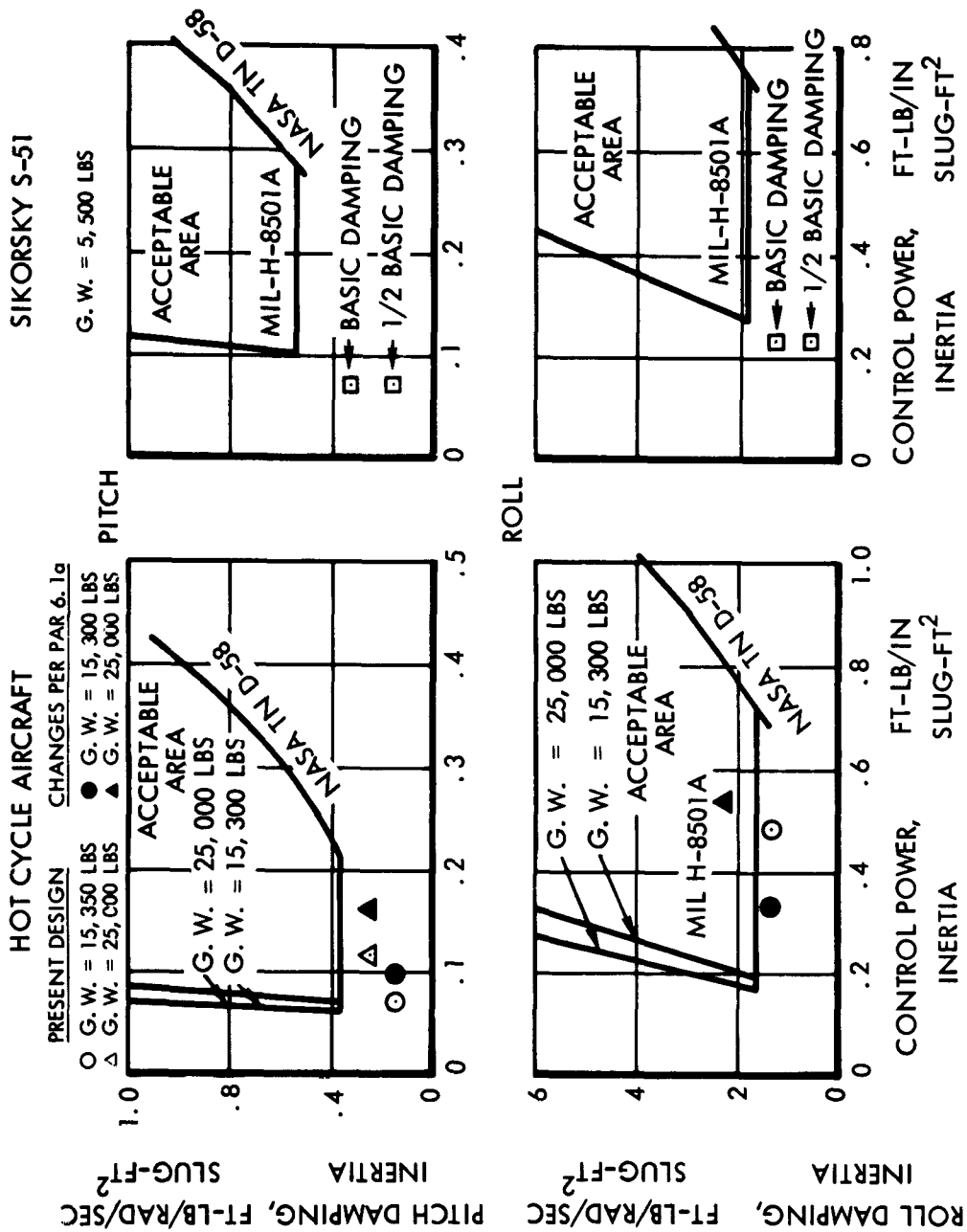
Figure 9. Handling Qualities Boundaries.

HOT CYCLE RESEARCH AIRCRAFT
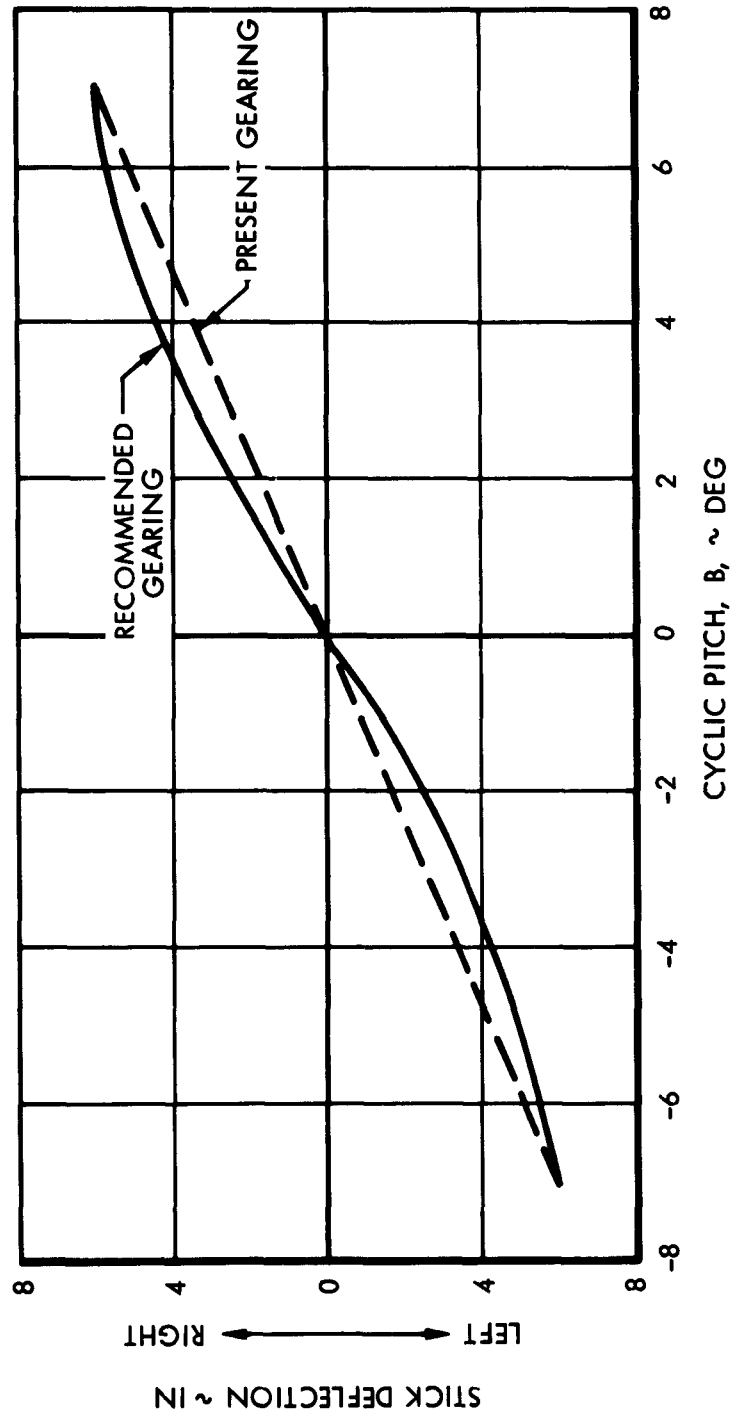
Figure 10. Lateral Cyclic Stick - Swashplate Gearing.

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be flown in the critical hover condition without undue pilot effort. This conclusion is based on the flight results of Reference 16.

Tests of Reference 16 were conducted using a Sikorsky S-51 to determine the effects of various combinations of damping and control power on helicopter handling characteristics for visual and instrument flights.

The basic damping and control power levels of the S-51 in pitch and roll are shown on Figure 9. It can be seen that the S-51, with its basic damping and control power, is not able to meet the handling requirements of Reference 15. However, the helicopter was flown under the critical handling requirements of instrument flight without noticeable difficulties. Further, the helicopter was flown under instrument flight with one-half the basic damping and original control power. Pilots' comments indicated that the handling characteristics were poorer than with the original damping and control power, but the helicopter could be flown without excessive pilot effort.

The hot cycle aircraft, with approximately 10 percent greater relative damping (MIL-H-8501A requirements equal to 100 percent) and 25 percent greater control power (for recommended increased control power, G. W. = 15,300 pounds), should have improved handling characteristics over those of the Sikorsky S-51 for the one-half damping case.

6.1.2   Handling Characteristics in Roll

The recommended lateral control power change of b. in Section 6.1, shown in Figure 10, has two purposes:

a. To reduce the control power for the initial ±50 percent stick travel, thus improving the hover handling characteristics in roll for the overload weight of 25,000 pounds as shown in Figure 9.

b. To retain the present high control power for flight maneuvers requiring large stick deflections. Thus, the requirements for the recommended nonlinear stick-swashplate gearing result.

The solid points shown on the roll curve of Figure 9 are with the modified control power. Results indicate that the handling characteristics of the hot cycle aircraft in roll should be substantially improved over those of the Sikorsky S-51 for the one-half damping case, and even for the full basic damping case.

6.1.3 Handling Characteristics in Yaw

The hot cycle aircraft meets the hovering yaw response characteristics per SPEC MIL-H-8501A as shown by the table below. (Based on an estimated maximum yaw thrust of ±300 pounds total.)

	SPEC MIL-H-8501A REQUIREMENTS (Degrees)	MODEL 385 (Degrees)
Yaw displacement at the end of 1 second per inch of pedal deflection	3.77	4.1
Yaw displacement at the end of 1 second for full pedal deflection	11.3	12.4
Yaw displacement at the end of 1 second from the most critical azimuth position during a 35-knot wind for full pedal deflection	3.77	5.2

The yaw angular velocity damping of the hot cycle aircraft is essentially zero. However, SPEC MIL-H-8501A states that the yaw angular velocity damping should preferably be at least $27 (I_z)^{0.7}$ feet-pounds/radius/second, thus, indicating that this is not a requirement.

It should also be noted that the absence of a tail rotor will make the research aircraft have less gust sensitivity in yaw. The yaw damping criterion was included in SPEC MIL-H-8501A chiefly because of the gust sensitivity in yaw of single-rotor helicopters with tail rotor. Therefore, this yaw damping criterion would not be a major consideration in the design of the hot cycle research aircraft. To emphasize this point, it is further noted that tandem helicopters generally do not meet this yaw damping criterion, nor do they have gust-sensitive tail rotors.

6.1.4 Summary

a. The present research aircraft, with the recommended changes in control powers, is expected to have improved handling characteristics in pitch and roll over those of the Sikorsky S-51 with one-half the original damping and original control power. Thus, the aircraft should be flyable in hover without excessive pilot effort.

b. The research aircraft damping characteristics in hover do not meet the damping requirements of Reference 15 in both pitch and roll except at the high gross weight of 25,000 pounds in roll.

c. Preliminary investigation has been made of the possibility of increasing the damping available to the aircraft by means of a mechanical or electronic device. It was found that by the introduction of damping, compliance with SPEC MIL-H-8501A is possible without introducing the stresses which accompany hub restraint when more control power is sought. Since a damping device is an additional item, the question of reliability and performance versus cost is introduced. The possibility of obtaining added damping with some simple device should be explored further.

6.2   ADVANTAGES AND DISADVANTAGES OF HUB RESTRAINT

It was pointed out in Sections 6.1.1 and 6.1.2 that the research aircraft will be controllable with the present control system configuration because it will have characteristics equal or superior to the Sikorsky S-51 helicopter with one-half damping. A substantial improvement in the degree of controllability and associated rotor damping available can be obtained if elastic hub tilt restraint is used. When the hub is restrained, the offset coning hinges that are part of the present rotor will start to function as offset flapping hinges as well. Then the large damping and control power contribution normally obtained from offset hinge rotors will be available for the research aircraft. As a result, the research aircraft would be considerably easier to fly, with reduced pilot effort and check-out time required. A possible way of obtaining the required hub restraint is discussed in Section 9.1.1.

Calculations were made to substantiate the statements above that indicate the characteristics of the research aircraft with elastic hub restraint will shift to the "acceptable" area, for all weight conditions, per requirements of SPEC MIL-H-8501A in both pitch and roll, as shown in Figure 9. At the overload weight condition of 25,000 pounds, the handling characteristics in roll are considered unacceptable per requirements of Reference 16. However, the handling qualities boundaries of Reference 16 are based on a specific helicopter (S-51) of GW = 5000 pounds and therefore may not reflect the effect of size on handling requirements. However, if flight test results indicate the requirements of Reference 16 are valid, the lateral control gearing of the research aircraft can simply be adjusted to give reduced control power, thus shifting the roll handling characteristics to the desirable area.

The single and major disadvantage of using hub restraint is that large one-per-rev stresses are introduced in the hub and shaft because of the bending moment introduced by the restrained rotor hub.

The amount of hub restraint considered above (15,000 inch-pounds of moment per degree of hub tilt) would approximately quadruple the stresses imposed on the shaft by the normal free-floating hub installation. This increase in stress is far too high for the strength of the existing shaft. A redesign of the shaft and some portions of the hub would be required. Since the research aircraft can be flown safely with the free-floating hub, and will be able to reach the speeds which are the objectives of the test program without hub restraint, it is planned to postpone any changes to the hub and shaft at this time to accommodate the loads which would be imposed by hub restraint. The suggested amount of hub restraint could be introduced at a later date, since it is still considered desirable to increase the rotor damping and control power.

6.3   FORWARD FLIGHT

6.3.1   Directional Stability

According to Paragraph 3.3.9 of SPEC MIL-H-8501A, "the helicopter shall possess positive, control fixed, directional stability and effective dihedral in both powered and autorotative flight at all speeds above 50 knots, $0.5 \, V_{Max}$, or the speed for maximum rate of climb, whichever is the lowest."

With this requirement in mind, an analysis was made of the research aircraft with the configuration shown in Figure 5. It was found that a vee-shaped tail, with a true area of 54 square feet and with its two halves inclined upward from the horizontal at 45 degrees, will produce the required directional stability.

6.3.2   Longitudinal Maneuver and Dynamic Stability

Paragraphs 3.2.11.1, 3.2.11.2, and 3.2.12 of MIL-H-8501A are concerned with longitudinal maneuver and dynamic stability. It was found that the vee-shaped 45-degree tail, sized at 54 square feet for adequate directional stability (see 6.3.1), will have a proper amount of projected horizontal area to meet adequately the longitudinal maneuver and dynamic stability requirements.

6.3.3   Stick Position Versus Speed

Paragraph 3.2.10 of SPEC MIL-H-8501A specifies that "the helicopter shall at forward speeds . . . possess positive, static longitudinal control force and control position stability with respect to speed." The stick force variation versus speed requirement will be met with a combination of trimmable springs. (The helicopter will use servos; therefore an artificial "feel" system will be necessary.)

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Stable control position stability is shown in Figures 11 and 12 for fixed stabilizer incidences for gross weights of 10,000 pounds and 25,000 pounds. This range of gross weights will cover all test cases from investigation of high autogyro speed at low gross weight to the alternate overload condition.

HOT CYCLE RESEARCH VEHICLE

HELICOPTER LEVEL FLIGHT
STANDARD SEA LEVEL
G.W. = 10,000 LBS

i_t = FIXED TAIL INCIDENCE

CG 6" FWD ROTOR ₵

CG 6" FWD ROTOR ₵

$i_t = -5°$

$i_t = 0°$

CYCLIC PITCH ANGLE, B, ~ DEG

FORWARD SPEED ~ KNOTS
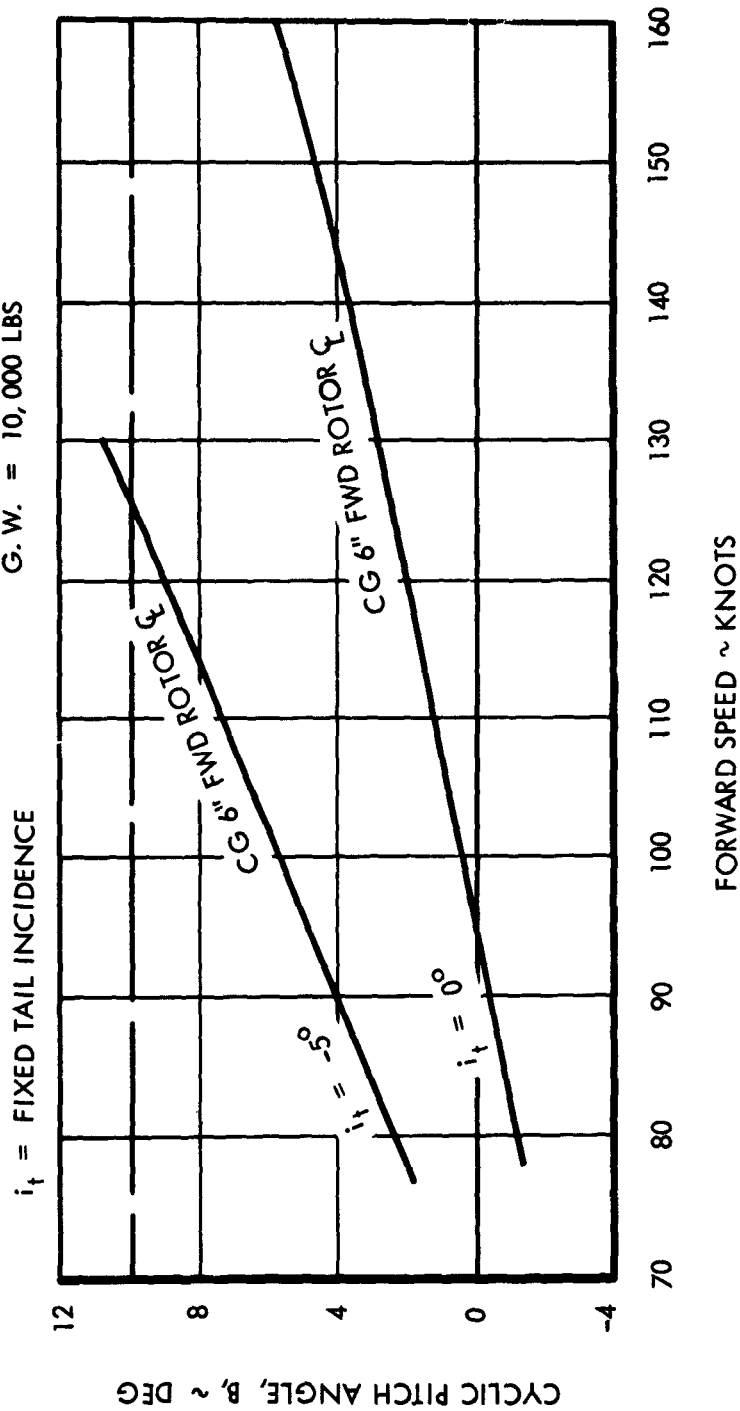
Figure 11. Trim Flight Cyclic Pitch Angle Vs. Speed (10,000 Pounds).

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HOT CYCLE RESEARCH VEHICLE

FIXED INCIDENCE TAIL

HELICOPTER LEVEL FLIGHT
STANDARD SEA LEVEL
G.W. = 25,000 LBS
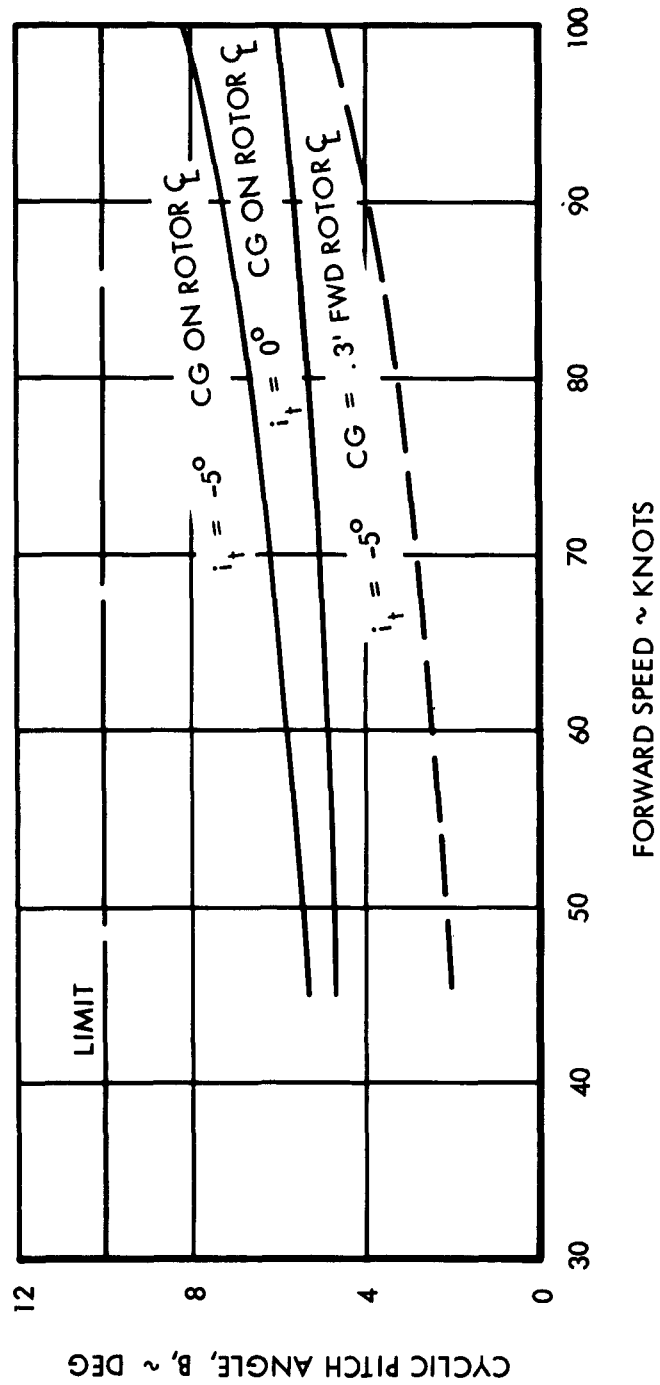
Figure 12.   Trim Flight Cyclic Pitch Angle Vs. Speed (25,000 Pounds).

7. STRUCTURAL CHARACTERISTICS

The major design criteria and structural characteristics of the research aircraft are given in this section. The design gross weight of the aircraft is 15, 300 pounds at a limit load factor of 2. 5. The design criteria also include design speeds as reported in Section 5. Additional design criteria and stress analysis for the rotor and upper control system are given in References 3, 17, 18 and 19.

7.1   STRUCTURAL DESIGN CRITERIA

7.1.1   Design Gross Weights

Design Gross Weight (maximum internal loading)	15, 300 pounds
Alternate Overload Gross Weight (maximum internal, plus external loading)	25, 500 pounds
Design Minimum Gross Weight	10, 000 pounds
Center of Gravity Limits (approximate)	$\mathcal{C}_L$ Rotor to 3. 3 inches fwd of $\mathcal{C}_L$ Rotor

7.1.2   Limit Flight Load Factors

Design Gross Weight $\qquad \begin{cases} +2.5 \\ -0.5 \text{ (internal loading only)} \end{cases}$

Design Minimum Gross Weight $\begin{cases} +3.0 \\ -0.75 \end{cases}$

Hovering alternate overload gross weight up to 25, 500 pounds (at a load factor of 1. 50) will be investigated.

7.1.3   Landing Criteria (limit) (ultimate = $\sqrt{1.5}$ x limit impact velocity)

As limited by strength of H-34 landing gear, n = 1. 75 at ground (approximately) at maximum landing gross weight of 15, 300 pounds and 6 feet per second impact velocity in combination with 0. 67 g rotor lift. Total helicopter acceleration at CG = 1. 75 + 0. 67 = 2. 42 g (limit).

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7.1.4   Design Speeds (sea level standard)

$$\left[\begin{array}{l} \text{G. W.} = 15,300 \text{ pounds} \\ \\ \text{Helicopter maximum level speed,} \\ V_H \text{ 145 knots} \end{array}\right] \left[\begin{array}{l} \text{G. W.} = 10,000 \text{ pounds} \\ \\ \\ 156 \text{ knots} \end{array}\right]$$

Helicopter design maximum dive speed,
$$V_{D_{15,300\#}} = 1.38 \times 145 = 200 \text{ knots}$$

$$V_{D_{10,000\#}} = 1.28 \times 156 = 200 \text{ knots}$$

Autogyro maximum level speed,   197 knots
V_H (faired configuration)

Autogyro design maximum dive speed,
$$V_{D_{10,000\#}} = 1.14 \times 197 = 225 \text{ knots}$$

7.1.5   Main Rotor Structural Criteria (blades and hub)

(Per Section I of Reference 3)

7.1.6   Upper Rotor Controls (including hydraulic actuators)

(Per Section I of Reference 3)

7.1.7   Lower Rotor Controls and Flight Controls

Limit Pilot Loads:

	Pilot To Stops (Pounds)	System Beyond Stops (Pounds)
Collective	100	60
Longitudinal Cyclic	100	60
Lateral Cyclic	67	40
Pedal (yaw or brakes)	130	78

Dual control loads are 75 percent of above values applied at each pilot station either in conjunction or in opposition.

7.1.8   Propulsion System Criteria (2 G. E. T64 type Gas
                                                Generators)

        Limit pressures and temperatures are taken from the T64
gas generator data given in References 11 and 12.

        A 1.33 limit factor is applied to advanced engine take-off
pressures of the referenced reports, or = 1.33 x 29.0 = 38.6 psig at
1184 degrees Fahrenheit. (The limit factor need not be applied on com-
bined flight loadings.) Target service life of hot parts under operating
conditions is 1,000 hours.

7.1.9   Tail Surface Loads (no ultimate factor)

        Symmetrical loading:

        C_n = 1.0 at V_D = 225 knots

        Chordwise distribution - c.p. 25 to 50 percent chord

        Unsymmetrical loading 0 — 100 percent distribution of above
        pressures

7.1.10   Ground Handling Loads (limit -   at 15, 300 pounds GW)

        a.   Mooring - 40-knot wind from any horizontal direction

        b.   Jacking - $\begin{cases} 2.0 \text{ g vertical} \\ 0.5 \text{ g fore or aft} \\ 0.5 \text{ g lateral} \end{cases}$

        c.   Hoisting - 2.0 g vertical

        d.   Towing - 0.25 g resultant horizontal ±30 degrees from
                        fore or aft

7.1.11   Crash Landing Ultimate Inertia Factors

        Separately: 10 g down, 10 g forward, 4 g side, 2 g up.

7.1.12   Ultimate Factor of Safety

        1.5 (except as otherwise specified herein)

7.2   PRELIMINARY STRUCTURAL DESIGN

7.2.1   Discussion

The preliminary structural design of the hot cycle research aircraft is primarily directed towards investigating the suitability of the hot cycle rotor system for operation in a flight vehicle and with the least development time and cost. Structural feasibility of the basic hot cycle rotor system has been demonstrated during over 60 hours of whirl testing. In keeping with the contract work statement and with the above objective, a number of nonoptimum materials, as well as numerous available components from other helicopters, are to be used in order to permit the flight research vehicle to be in operation at an early date and with the least development costs. Thus, it is realized that the optimum weight reduction potential for a hot cycle service helicopter will not be achieved in this research vehicle and that further structural design refinement could considerably enhance the payload and performance capability of a similar production aircraft. Maximum use will be made of whirl tower components such as upper control system, hub and shaft components, ducting, and the rotor blades. Available structural components which are utilized from other helicopters include the Hughes HO-6 cockpit and enclosure, including the lower flight control system from pilot to servo valves, and Sikorsky H-34 main and tail landing gear assemblies.

Structural design of the research aircraft matches the whirl tower proven hot cycle rotor system to a simple, efficient, aerodynamically clean aluminum alloy semimonocoque airframe. From the forward portion of the airframe which is based on Hughes HO-6 tooling, a constant center section extends aft blending into a conical aft boom section supporting a butterfly tail, yaw control jets, and the tail landing gear at its aft extremity. In order to save weight, improve ducting, and to simplify structure and reduce drag, the engine nacelles and diverter valves (as well as the main landing gear struts) are supported on lateral pylons disposed at each side of the upper fuselage and mounted at the same fuselage frame stations as the main rotor pylon.

The structural design of the hot cycle research aircraft emphasizes simplicity, not only to reduce development and tooling costs but also to improve reliability and safety. The airframe, in general, is of semimonocoque aluminum alloy construction and therefore has sufficient redundancy of members to be considered fail-safe. Selection of structural materials for the airframe is generally conservative in order to avoid such problems as stress corrosion cracking, plating embrittlement, and fatigue. The three-bladed, gimbal-mounted, hot cycle rotor system is expected to impose comparatively low vibration levels on the remainder of the airframe; thus, fatigue problems are not anticipated

for the research aircraft airframe structure. Mounting of the engine nacelles, diverter valves, hot gas ducting, and rotor system on external faired pylons minimizes the danger of fire and permits the majority of airframe components to operate at ambient temperatures acceptable for aluminum alloy construction. Hot gas components will make use of high-temperature materials such as Rene' 41 and Inconel X, that have been proven reliable during whirl tower rotor operation. No new design or fabrication techniques are therefore required for these hot components. Removal of the hot gas overboard exhaust from the immediate airframe area also minimizes problems due to sonic fatigue of the sheet-metal structure. Minor modifications in the hub, blades, and upper controls will be in the nature of design improvements to assure even greater reliability and safety in these components when used on the research aircraft.

Hub restraint (Reference Section 6.2) was considered and found to apply excessive stresses on the existing shaft. Therefore, further design consideration is not given at this time. However, hub restraint can be incorporated at a future date by redesign to strengthen the affected components.

7.2.2   Airframe Design

a. Fuselage. The main rotor is supported from the fuselage through a main rotor pylon structure. Members transmit loads from the upper and lower main rotor shaft bearing housings directly to four major hard points in the upper fuselage monocoque structure. The existing whirl tower pylon structure is not adaptable to the dual T64 engine arrangement. The lateral pylons on each side of the upper fuselage support nacelles with engines and diverter valves as well as the main landing gear strut. They are built into the same major fuselage frames and longerons which support the rotor pylon, thus eliminating duplication and shortening the load paths. The nacelle-pylon arrangement provides simplicity of structure, maximum accessibility, excellent duct routing, and minimum airframe weight.

An aluminum alloy semimonocoque fuselage structure was chosen for the hot cycle research vehicle because it provided simple tooling, high strength-weight efficiency, and minimum development time and cost. The forward cockpit and enclosure is based on the Hughes HO-6 tooling with member gages increased where necessary. The fuselage continues in constant cross section throughout the central portion of the fuselage. The aft fuselage, or boom, is a stiffened circular cone providing support for the butterfly tail, the yaw control jets, and the tail landing gear. Fuel is carried in the central fuselage section. The stiffened aluminum alloy construction is shown in Section 7.2.3 to be near optimum for the imposed loadings. Aluminum alloys (2024) will

generally be used for the fuselage monocoque structure to simplify fabrication, reduce cost, and minimize development problems.

Crashworthiness features of the Hughes HO-6 cockpit enclosure, such as integral seat and safety harness support structure, will be utilized in the research aircraft. Major fuselage members supporting the rotor and the nacelles are designed for a crash load factor of 10 g - or 2-1/2 times minimum FAA requirements (Reference Section 7. 1). Structure supporting fuel tankage is designed with due consideration for good crashworthy and fireproof characteristics. Provision for external cargo support directly from the rotor thrust bearing housing and pylon is provided, thus relieving the fuselage structure of this additional loading. An opening through the lower fuselage provides adequate clearance for swinging of the cargo.

b. Nacelle and Lateral Support Pylons. In Figure 13, each General Electric T64 gas generator is shown supported on a six-component mounting system which isolates thermal and structural strains from the engines. Each diverter valve and ducting section is supported on an additional four-component mounting system which supports all major gas pressure loads, including one engine inoperative, and dead-weight loads from the diverter section, leaving only two light positioning load components to be supported at the slip-joint to the engine outlet. Slip-jointed and articulated attachments isolate thermal and structural strains between the diverter section and the remaining ducting to the rotor.

A two-spar, stressed-skin, airfoil-shaped pylon attaches the nacelle to the main fuselage structure at the same main frame stations to which the main rotor pylon attaches. The lateral pylon structure also supports the main landing gear shock strut loads, to even further avoid duplication of fuselage members. Landing inertia loads from the rotor and propulsion system thus bypass the fuselage entirely.

c. Landing Gear. Available Sikorsky H-34 landing gear components are used for both the main and the tail landing gear of the hot cycle research aircraft in order to reduce development time and cost. The main oleo strut geometry is rearranged to a more nearly vertical position for attachment to the nacelle pylons. The Sikorsky landing gear was originally designed for 8 feet per second contact velocity and 11,400 pounds landing weight; however, modifications will be made as necessary to provide for 6 feet per second contact velocity at 15,300 pounds gross weight for the hot cycle research vehicle. Since chordwise natural frequencies of the hot cycle rotor system are above all normal rotor speeds, ground resonance will not become a problem due to landing gear modifications.

SEALED BALL-SLIP JOINTS
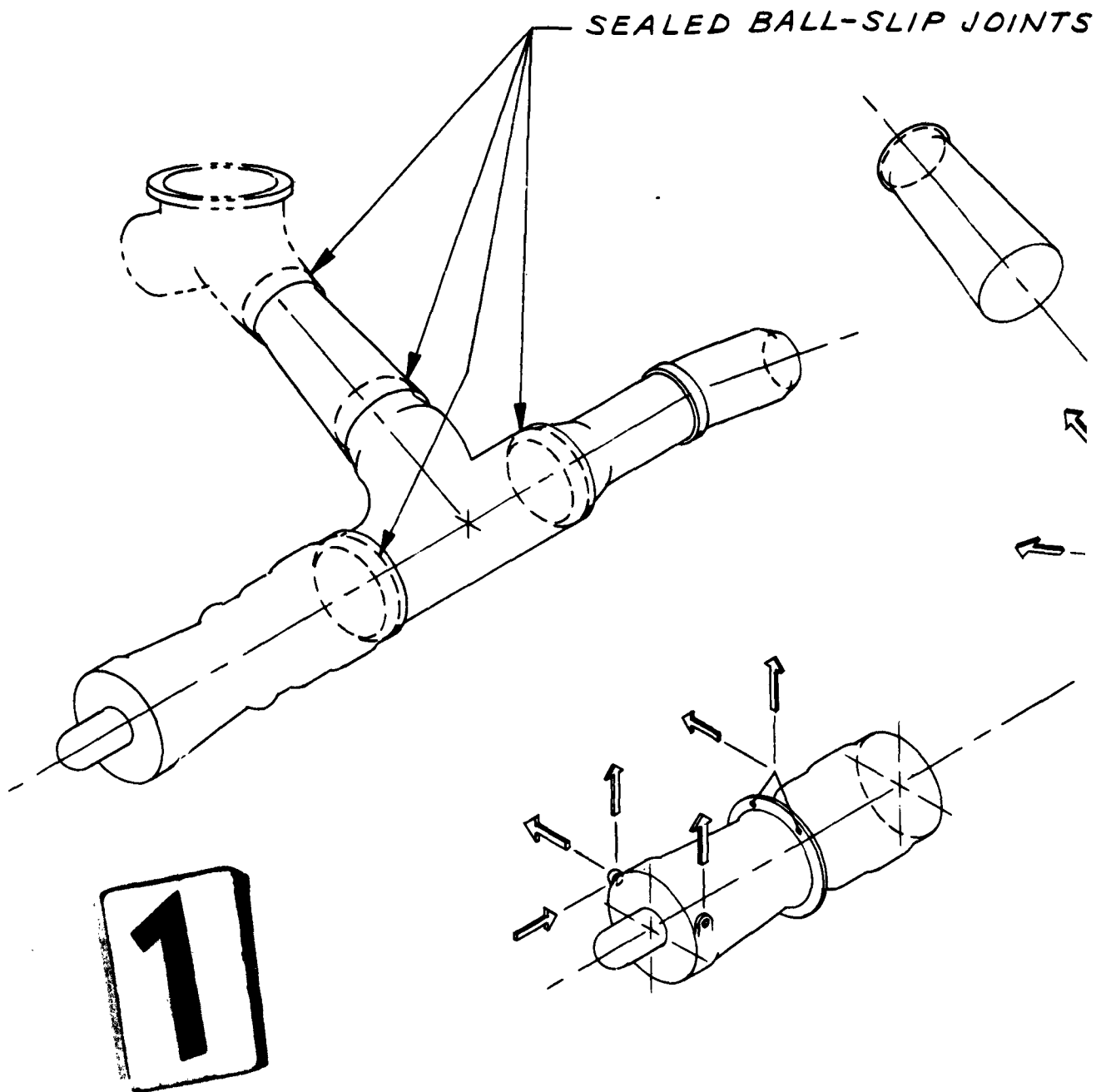
FIG. 13, POWER PLANT MOUN

Figure 13. Power Plant Mounting System.

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"A" RADIAL RESTRAINT AT SLIP JOINTS
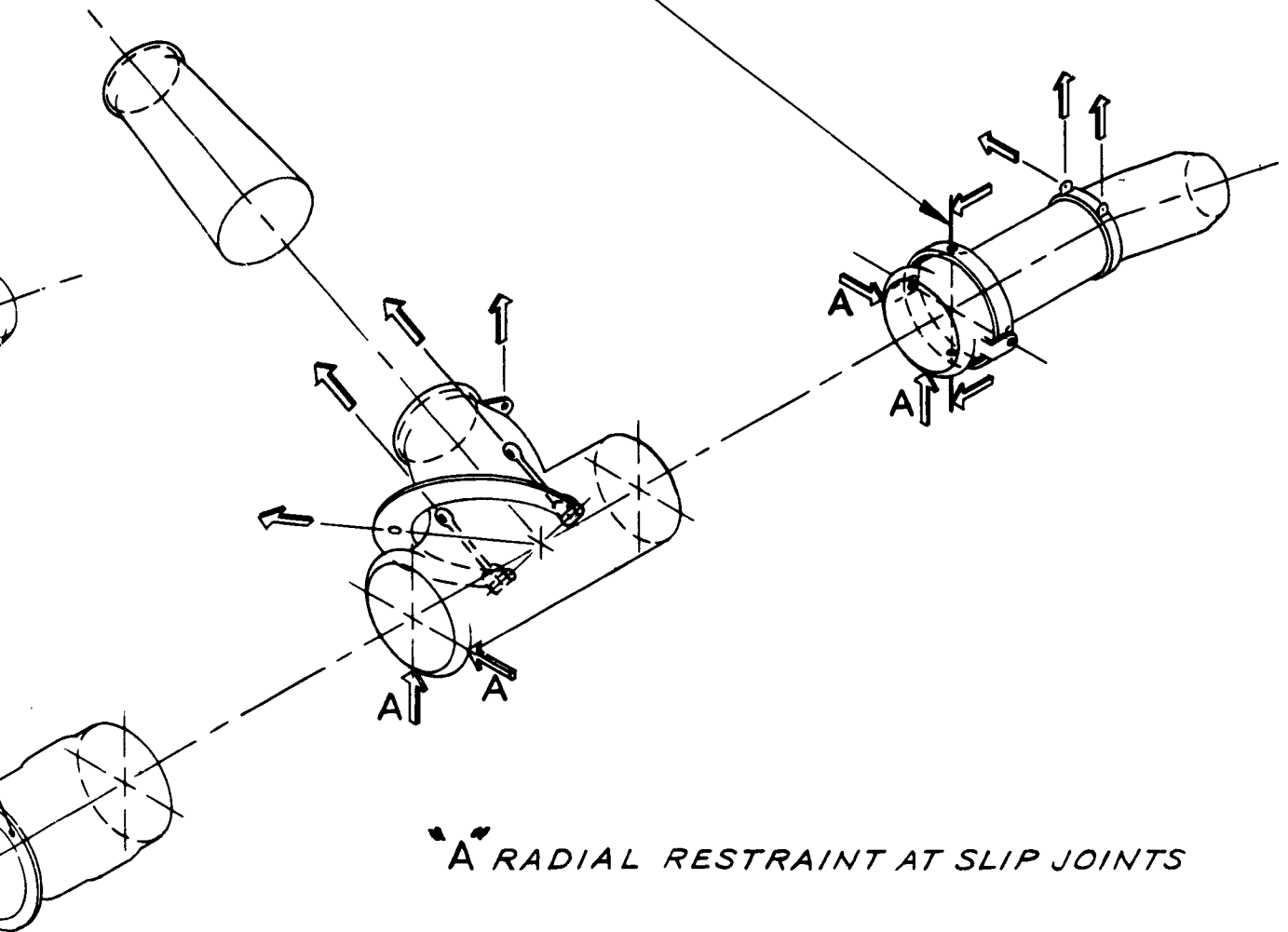
ER PLANT MOUNTING SYSTEM

2

d. Stabilizing and Control Components. Yaw and pitching stability of the helicopter are provided by a butterfly tail surface arrangement at the end of the fuselage boom. Simple two-spar, semi-monocoque structure is provided, utilizing a beaded outer skin and a minimum of ribs or internal stiffeners in order to simplify fabrication and tooling and to reduce cost. Yaw control jets on each side of the aft tail cone are also provided for hovering yaw control.

e. Flight Control Systems. The cockpit controls are HO-6 components. Other available HO-6 components are utilized whenever possible in the lower flight controls from the cockpit to the servo valves. Hydraulic valves and cylinders are also available items (reference Figure 14).

7.2.3   Basic Loads

The following typical preliminary basic load calculations are intended to substantiate values shown in Section 7.1, Structural Design Criteria, and to indicate the method of obtaining critical loadings for use in preliminary structural design of the airframe.

a. Landing Loads

(1) Sikorsky H-34 Landing Gear

Limit Drop:

Landing Weight = 11,400 pounds

Rotor Lift = 1/2 W

Contact Velocity = 8 feet per second

Assumed tire efficiency at 30 percent and
6.5-inch deflection

Assumed oleo efficiency at 70 percent and
9.5-inch vertical axle travel (reference Figure 15)

$$\frac{12 \times (8)^2}{2 \times 32.2} + (9.5 + 6.5) - 1/2(9.5 + 6.5)$$

$$= n(0.70 \times 9.5 + 0.30 \times 6.5)$$

$$n = \frac{19.9}{8.6} = \underline{2.32 \, g} \text{ (limit ground load factor)}$$

Alternate Computation:

Ultimate Helicopter Drop Load Factor = 4.0 g (per verbal
information)

-0.5 g (rotor lift)

= 3.5 g (ground)

Limit Ground Load Factor = $\frac{3.5}{1.5}$ = 2.33 g

∴ H-34 landing gear will support a total limit ground load
≥ 11.400 x 2.33
≥ 26,600 pounds

(2) Hot Cycle Research Aircraft Landing Gear

Because of the ability to maintain flight with one engine
inoperative, the criteria of 6 feet per second contact velocity in com-
bination with 2/3 W rotor lift (reference MIL-S-8698 (A.S.G.), Para-
graph 3.4.2) is considered acceptable as a limit condition for the hot
cycle research aircraft

$\frac{12 \times (6)^2}{2 \times 32.2}$ + (9.5 + 6.5) - 2/3 (9.5 + 6.5) = + n (.70 x 9.5 + .30 x 6.5)
+ 6.7 + 5.3 = 8.6 n

n = $\frac{12.0}{8.6}$ = 1.40 g (limit ground load factor)

Hot cycle research aircraft design landing weight = 15,300 pounds.

∴ Total hot cycle limit ground load = 15,300 x 1.40 = 21,300 pounds,
and hot cycle research aircraft landing gear will safely withstand

a limit ground load factor = $\frac{26,600}{21,300}$ x 1.40

= 1.75 g (use) (conservative)

(3) Tail Wheel Loads

Due to the distance of the tail landing gear from the heli-
copter center of gravity in relation to the main gear and in relation to
the helicopter radius of gyration, the translational impact is seen to be
somewhat more critical than a tail first contact (as shown in the follow-
ing calculations).

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Figure 14. Inboard Profile Hot Cycle Research Aircraft.

75

2
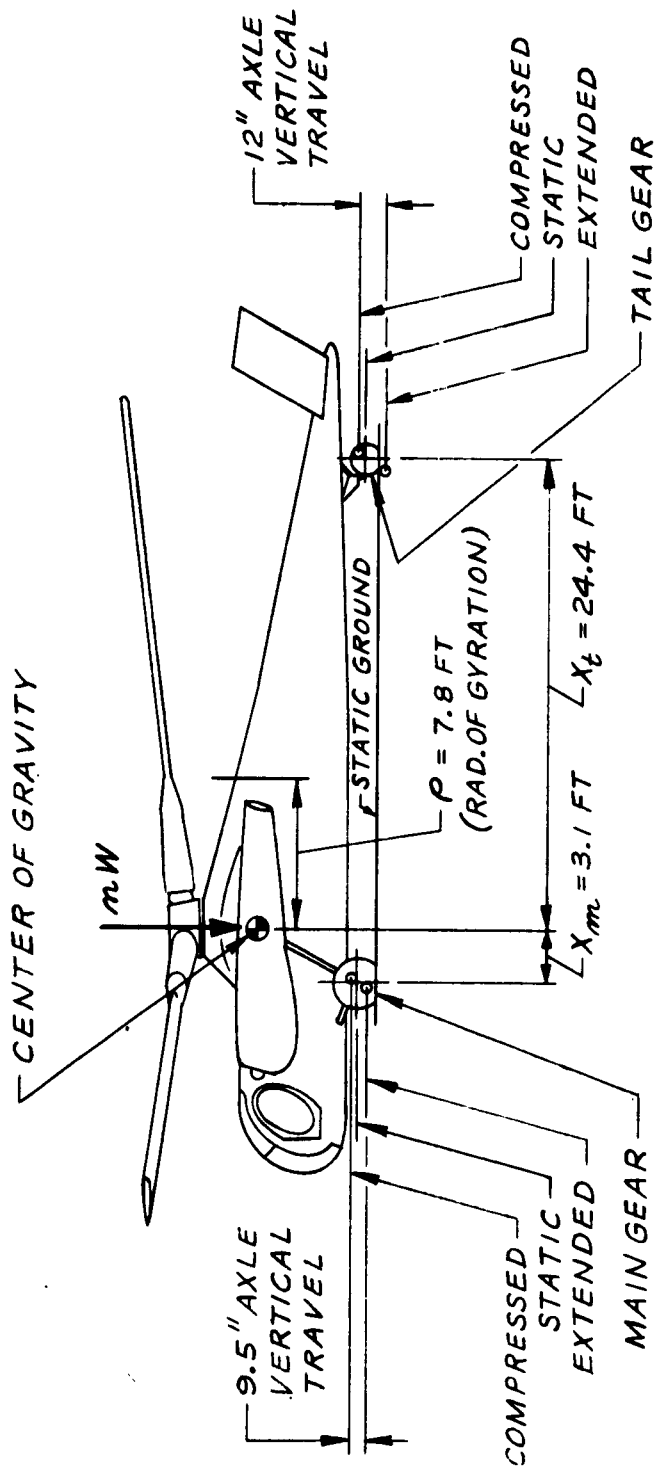
Figure 15.   Landing Gear Data

(REF. FIG. 5)

Figure 15.   Landing Gear Data.

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Effective mass ratio (tail first) (reference Figure 15)

$$\frac{W_e}{W} = \frac{1}{1 + \left(\frac{x_t}{\rho}\right)^2} = \frac{1}{1 + \left(\frac{24.4}{7.8}\right)^2} = \underline{0.0926}$$

Effective mass ratio (3 point) $\quad \dfrac{W_e}{W} = \dfrac{x_m}{x_m + x_t} = \dfrac{3.1}{27.5} = \underline{0.113}$ (use)

Since the total tail gear vertical axle travel due to oleo travel = 12.0 inches (reference Figure 15) and the tire deflection at maximum load $\cong$ 4.0 inches (for a total travel of the effective mass approximately equal that of the main gear), the same ground load factors as developed for the main gear (reference item a. above) will be used for the tail gear.

Tail Gear Load Factor = 1.5 x 1.75 = 2.62 g ultimate at 11.3 percent effective mass.

Design Maximum Landing Weight = 15,300 pounds

Tail Wheel Load = 0.113 x 15,300 x 2.62

= 4,530 pounds (ultimate)

(Less critical than tail surface load of item b. below)

b. Tail Surface Load

C_N Maximum = 1.0 at 225 knots (maximum design dive speed)

Total Tail Area (horizontal projection) = 0.707 x 54
= 38.1 square feet

Maximum Vertical Load = 1.0 x 0.00256(1.15 x 225)^2 x 38.1

= 6,500 pounds (ultimate, up or down)

c. Engine - Diverter Valve Loads. Direction and character of support loads are shown in Figure 13. An ultimate factor of safety of 1.5 will be maintained. Target service life shall be 1,000 hours at operating temperatures and pressure.

7.2.4 Weight - Strength Optimization of Aft Fuselage Structure

Weight-strength ratios can be conveniently evaluated by the parameter

$$\frac{A}{M^{2/3}}$$

Where A is the total cross sectional area of a 2024 aluminum alloy stiffened cylindrical shell, and M is the bending moment (reference 20).

$$\frac{A}{M^{2/3}} = \frac{4\left(\frac{M}{D^3}\right)^{1/3}}{f_c} + \frac{\pi C_2}{\left(\frac{M}{D^3}\right)^{1/6}\left(\frac{L}{D}\right)^{3/2}}$$

where

$$C_f = \frac{1}{16,000}$$

$$K_4 = 5.24$$

A plot of this parameter is obtained for L/D of 0.10 and 0.20 and is shown in Figure 16.

The following preliminary design calculations were performed to indicate the most nearly optimum structural arrangement for the aft fuselage and to assist in substantiation of weight estimates. Similar calculations (not included) were also performed for the nacelle and rotor pylon structure to assist weight estimation. Structural index (Reference 20) and beam load intensity are calculated for four typical aft fuselage stations.

7.2.5 Fuselage Design Calculations

a. Section - Station 580:

Circular Diameter = 25.5 inches

Moment = (651.5 - 580) x 6,500 = 466,000 inch-pounds
(ultimate)

Structural Index, $\frac{M}{D^3}^{1/3}$ = $\frac{77.7}{25.5}$ = 3.04 , and

Load Intensity = $\frac{466,000}{0.785(25.5)^2}$ = 910 pounds per inch
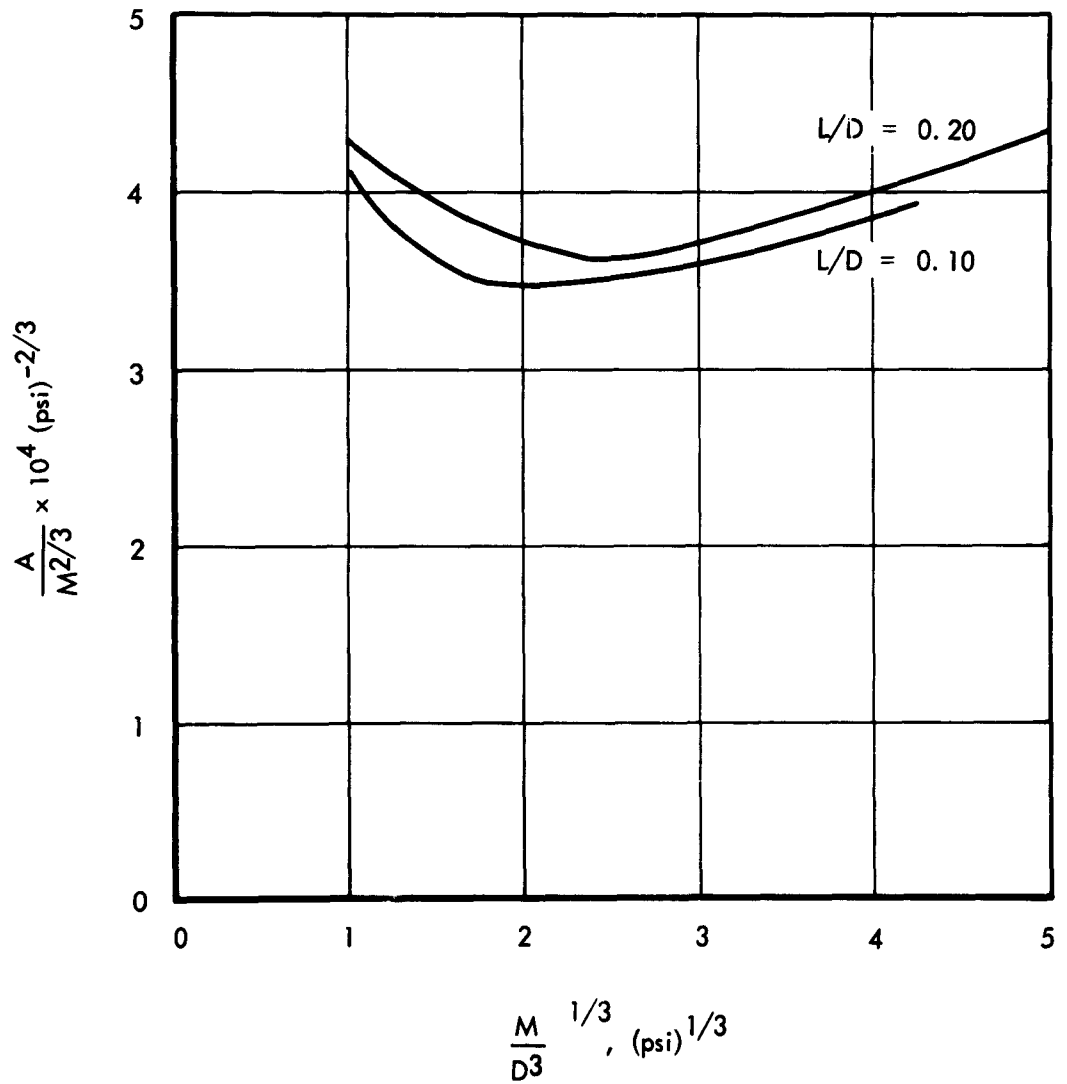
Figure 16. Optimum Fuselage Parameters.

b. Section - Station 468.5:

Circular Diameter - 49.5 inches

Moment = (651.5 - 468.5) x 6,500

= 1,190,000 inch-pounds (ultimate)

Structural Index $\left(\dfrac{M}{D^3}\right)^{1/3}$ = $\dfrac{106}{49.5}$ = 2.14

Load Intensity = $\dfrac{1,190,000}{0.785\,(49.5)^2}$ = 618 pounds per inch

c. Section - Station 408.5:

Approximate Circular Diameter = 60 inches

Moment = (651.5 - 408.5) x 6,500 = 1,580,000

inch-pounds (ultimate)

Structural Index $\left(\dfrac{M}{D^3}\right)^{1/3}$ = $\dfrac{117}{60}$ = 1.94

Load Intensity = $\dfrac{1,580,000}{0.785\,(60)^2}$ = 558 pounds per inch

d. Section - Station 617.5:

Circular Diameter = 17.7 inches

Moment = (651.5 - 617.5) x 6,500

= 221,000 inch-pounds (ultimate)

Structural Index $\left(\dfrac{M}{D^3}\right)^{1/3}$ = $\dfrac{60.5}{17.7}$ = 3.42

Load Intensity = $\dfrac{221,000}{0.785\,(17.7)^2}$ = 893 pounds per inch

7.2.6   Representative Fuselage Skin Stringer Panels

The following fuselage panel design calculations are based on calculated panel load intensities from pages 79 and 81. Stringer and skin sizes are derived for the representative fuselage stations.

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a. At 5 inches stringer spacing and approximately 900 pounds per inch (represents stations 580 and 617. 5):

Load per Stiffener = 900 x 5 = 4, 500 pounds (ultimate)

Allowable Stress, F_c = 29,000 psi (estimated for 2024 aluminum alloy and

$\frac{L^1}{\rho} \cong 48$ at 20-inch frame spacing)

Total Required Effective Area = $\frac{4,500}{29,000}$ = 0.153 square inch

Area of 40 t effective skin at 0.032 inch = 40 $(0.032)^2$ = 0.041 square inch

Stiffener Area Required = 0.153 - 0.041 = 0.112 square inch

Stiffener Area = 2.8 (developed length from sketch) x 0.040 = 0.112 square inch

From Sketch, $\rho \cong 0.30$ x 1.13 = 0.34 inch

For 20-inch frame spacing,

$L^1 = \frac{L}{\sqrt{1.5}} = \frac{20}{1.225}$

$= 16.3$ inches

and $\frac{L^1}{\rho} = \frac{16.3}{0.34} = 48$

Column Allowable Stress
= 29,000 psi

TYPICAL STIFFENER
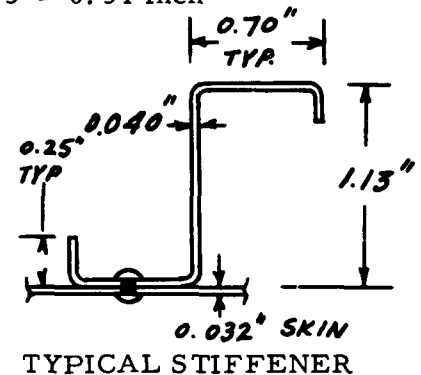
b. At 5-inches stringer spacing and approximately 600 pounds per inch (represents stations 468.5 and 408.5):

Load per Stiffener = 600 x 5 = 3000 pounds (ultimate)

Allowable Stress, F_c = 26,000 psi (estimated)

Total Required Effective Area = $\frac{3,000}{26,000}$ = 0.114 square inch

$$\text{Area of 40 t Effective } \underline{\text{Skin at 0.025 inch}} = 40 \ (0.025)^2$$
$$= 0.025 \text{ square inch}$$

$$\text{Stiffener Area Required} = 0.114 - 0.025 = 0.089 \text{ square inch}$$

Stiffener Area = 2.8 x 0.032 = $\underline{0.089 \text{ square inch}}$ for 0.032 inch stringer similar to sketch above.

From the curve of average areas for 2024 aluminum alloy stiffened cylinders (frames and stringers) (Reference Figure 16), it may be seen that the aft fuselage structural indices of approximately 2 to 3 (reference pages 79 and 81) are near optimum for a 24ST stiffened cylindrical shell. The average frame spacing of 20 inches is 3 times the optimum (since $\dfrac{L}{D} = \dfrac{20}{60} = 0.33$ vs. optimum = 0.115); however, the weight increase is small due to the too few frames while the cost and time savings are great. It is concluded that the fuselage design for the hot cycle research aircraft maintains reasonable strength-weight efficiency, while offering ease of fabrication through simplicity, together with low development risk through a straightforward design approach.

8. DYNAMIC CHARACTERISTICS

8.1   ROTOR DYNAMICS

The rotor which will be used on the Research Aircraft is essentially the one which was used in the whirl tests reported in Reference 10 and for which the dynamic characteristics were reported in Reference 21. The flapwise and chordwise natural frequencies of the whirl test rotor are given here in Figure 17, which is reproduced from Figure 1 of Reference 21.

It is seen that no chordwise or flapwise resonances (circled points) occur in the operating range of the rotor. The closest possible resonances to the operating range occur between chordwise pinned first mode and 6/rev, and chordwise cantilever second mode and 7/rev. Both of these resonances will occur at about 90 percent of the minimum operating RPM. This 10-percent margin is considered adequate, considering that the system has some inherent damping.

The only design change contemplated in the rotor at this time would involve increasing the chordwise stiffness of the first cantilever chordwise mode to raise its natural frequency about 10 percent further away from one per rev than shown in Figure 17 in the operating range. This particular mode is more removed from actual resonance than the two other modes mentioned above, but there is such a strong one-per-rev forcing function that the first mode responds fairly strongly. This response can be reduced, and blade life increased, by increased chordwise stiffness. The forcing functions at two per rev, three per rev, etc., which excite higher modes, are of much smaller magnitude, leading to acceptable stresses in the operating range.

8.2   FUSELAGE VERTICAL AND LATERAL NATURAL FREQUENCIES

Because of the rather long and slender proportions of the fuselage of the research aircraft, a brief review was made of the probable natural frequencies of the fuselage for both vertical and lateral motions. No generalized data could be found that would be applicable for such a fuselage, and the preliminary nature of this study precluded a detailed analysis of the fuselage mass and stiffness distribution to compute natural frequencies. However, some unpublished data were available on the vertical and lateral natural frequencies of the fuselage of another single-rotor helicopter (the H-34) which is approximately the same length as that for the research aircraft. This reference fuselage
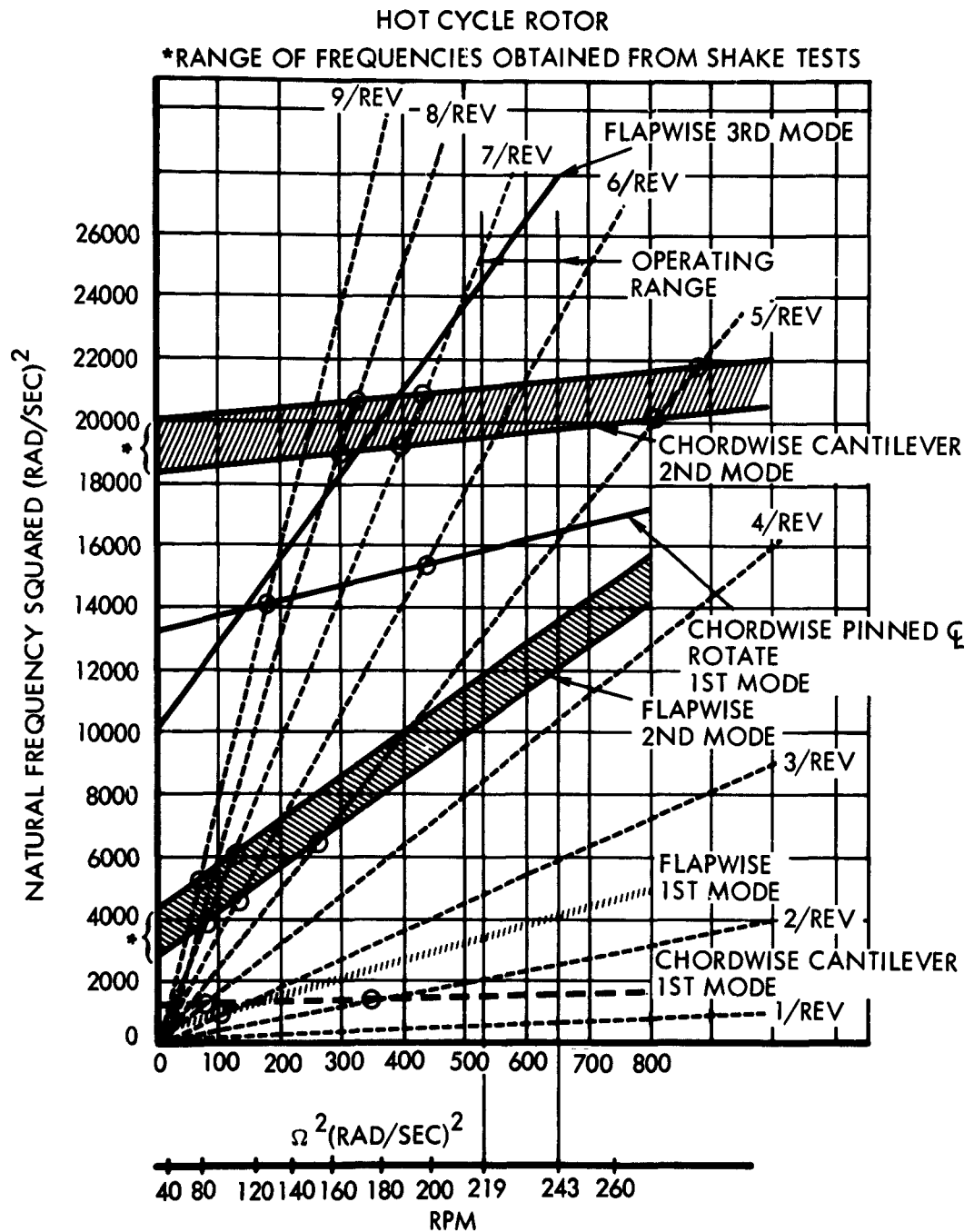
Figure 17. Predicted Dynamic Characteristics.

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was substantially deeper than that proposed for the research aircraft. However, the reference fuselage was also a heavier one, with a heavier concentration of mass at its aft end. As a result, it was felt that the frequencies of the reference helicopter fuselage would reasonably represent what might be found for the research aircraft. The vertical natural frequency of the reference fuselage was about 400 cycles per minute; the lateral natural frequency was about 350 cycles per minute.

The rotor for the research aircraft will operate at 243 rpm. This will lead to a one-per-rev input forcing function from the rotor due to blade unbalance, unequal moments, etc., of 243 cpm. Because of the three blades, the first level of aerodynamic inputs from all three blades operating equally will be 729 cpm. If the assumed analogy of the frequencies of the fuselage of research aircraft to that of the reference fuselage is valid, then it could be expected that the most probable inputs from the rotor (one per rev and three per rev) will bracket the probable vertical and lateral natural frequencies of the fuselage. As a result, the fuselage could be expected to shake very little in response to vibration forces caused by the rotor. Actual mass and stiffness distributions will be required to establish a more reliable estimate of these frequencies. Shake tests should be performed prior to flight to substantiate the correctness of the foregoing tentative conclusions that no major fuselage response exists for normal operation.

9. PRELIMINARY DESIGN OF MAJOR SYSTEMS

This section contains preliminary information on the design of all major systems and subsystems of the complete research aircraft shown in Figure 5. Drawings, both layout and detailed, which were prepared in the performance of this study, are presented herein. A brief description of the major systems is also given.

9.1 ROTOR SYSTEM

The general configuration of the rotor system is shown in Figures 5 and 18. This rotor is to be the same rotor which was successfully tested in the whirl tests reported in Reference 10, with the exception of the changes listed below. It is constructed with two primary steel spars which carry centrifugal and bending loads. The hot gases are carried to the blade tip nozzles through two ducts made of a succession of slip-jointed boxes of Rene' 41 alloy, a high-temperature, high nickel-based alloy. The slip-jointed boxes are attached to the two spars by chordwise ribs, to which the external aerodynamic skin is attached. A nose fairing is attached over the leading edge spar. A set of aluminum trailing edge pockets are attached around the rear spar to complete the aerodynamic contour of the NACA 0018 airfoil.

The blades are each attached to the rotor hub with two bundles of thin steel straps. The rotor hub is gimbal mounted in a free-floating manner to the rotor mast. The blade pitch arms are actuated from centrally located control rods which are controlled from a swashplate located beneath the rotor hub.

The following changes will be made to the existing rotor system as a result of the previous 60-hour whirl test and component tests.

    a. Incorporate blade tip closure valves in accordance with the detail design reported in Reference 24. This will permit single engine operation in the research aircraft.

    b. Provide reinforcement of hub gimbal lugs to increase strength of gimbal system for in-plane loads.

    c. Add gimbal lug thrust bearings to provide direct load path for in-plane loads.

    d. Provide blade spars of steel in place of titanium to eliminate excessive notch sensitivity of titanium alloy.

Figure 18. Hot Cycle Rotor.

e. Provide increased natural frequency in chordwise bending of blades to increase spread from operating frequency.

f. Provide reinforced articulate duct clamped joints to eliminate leakage during severe maneuvers.

g. Reduce weight of swashplate to improve structural efficiency and to provide for new hydraulic actuator positioning.

h. Provide lightweight lower Y-duct and upper rotating triduct to improve structural efficiency and to reduce stresses during transient power conditions.

9.1.1 Hub Restraint

Hub restraint will not be used initially on the research aircraft. However, in anticipation that hub restraint may eventually be used, a suggested means of providing rotor hub elastic restraint in a simple, inexpensive manner is shown on Figure 19. The restraint consists of a ring of BUNA-N rubber, which is compressed whenever the hub tilts with respect to the shaft. Although this rubber type of restraint is subject to high hysteresis under large deflections, an effective hub deflection of 3 degrees with the moment gradient calculated to meet the requirements of MIL-H-8501A can be achieved with the BUNA-N rubber, which will be adequate for initial testing of a hub restraint system. However, a maximum of 6-degree hub deflection will be necessary for a full test program of a restrained hub. For this much hub tilt, a different method of restraint is required and a "liquid spring" seems promising. Further work will be required to obtain maximum hub restraint if it is decided to install this type of device. The rework of rotor hub and shaft mentioned in Section 6.2 will be required, of course, to maintain stresses at acceptable levels.

9.2 PROPULSION SYSTEM AND NACELLES

The power plant installation for this research aircraft features good inspection and maintenance access, convenient engine and diverter valve removal, good fire detection, protection, and prevention, and aerodynamically clean inlet conditions. The major components of the propulsion system, such as, engine, diverter valves, mounting, ducting, and nacelle structure, are shown in Figures 20 and 21. Additional information on engine controls, cooling system, fuel system, etc., is given below.

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9.2.1 Engines

The engines are gas generator versions of the General Electric T64. No single drawing is available from General Electric which shows the complete gas generator. However, the composite drawing of the engine shown in Figures 20 and 21 was prepared according to instructions from General Electric and represents the engine gas generator configuration that will be furnished by General Electric for the research aircraft. The front of the engine from the compressor inlet to the beginning of the hot section is taken from Reference 22, omitting the power output shaft. The hot section of the engine, which consists of a two-stage turbine sufficient to drive the gas generator only (instead of the usual four-stage, two-piece turboshaft turbine), is taken from Reference 23, and includes components that were used during development testing.

9.2.2 Diverter Valves

General Electric J85 diverter valves are used. The dimensions are given in Reference 7. Because the inlet diameter of the diverter valves is about 3 inches smaller in diameter than the exit from the gas generator, a small transition section is required between the gas generator and the diverter valve. This short-piece transition ducting is shown on Figure 21. This same transition piece also includes a ball-type seal which provides for thermal expansion and/or misalignment between the engine and diverter valve. The diverter valve's position is controlled from the cockpit by hydraulic system actuation.

9.2.3 Hot Gas Ducting

Figures 20 and 21 show the two ducts which carry hot gas either to the "Y" duct at the rotor hub for helicopter operation or to the rearward facing nozzles for autogyro operation, depending on the position of the diverter valves. Both of these ducts have flexible joints which allow for thermal expansion or misalignment.

It should be noted that the autogyro nozzles also serve as engine exhausts when starting the engines on the ground before it is desired to start the rotor turning.

9.2.4 Engine Controls

The engine fuel control utilizes two airframe supplied linkage connections to two concentric control input shafts. The power control shaft is linked to a cockpit power control for engine starting, acceleration, shutdown, and rotor speed setting. The load signal shaft is linked to collective pitch. For normal flight operation the rotor speed governor acts as a droop governor to regulate gas generator power
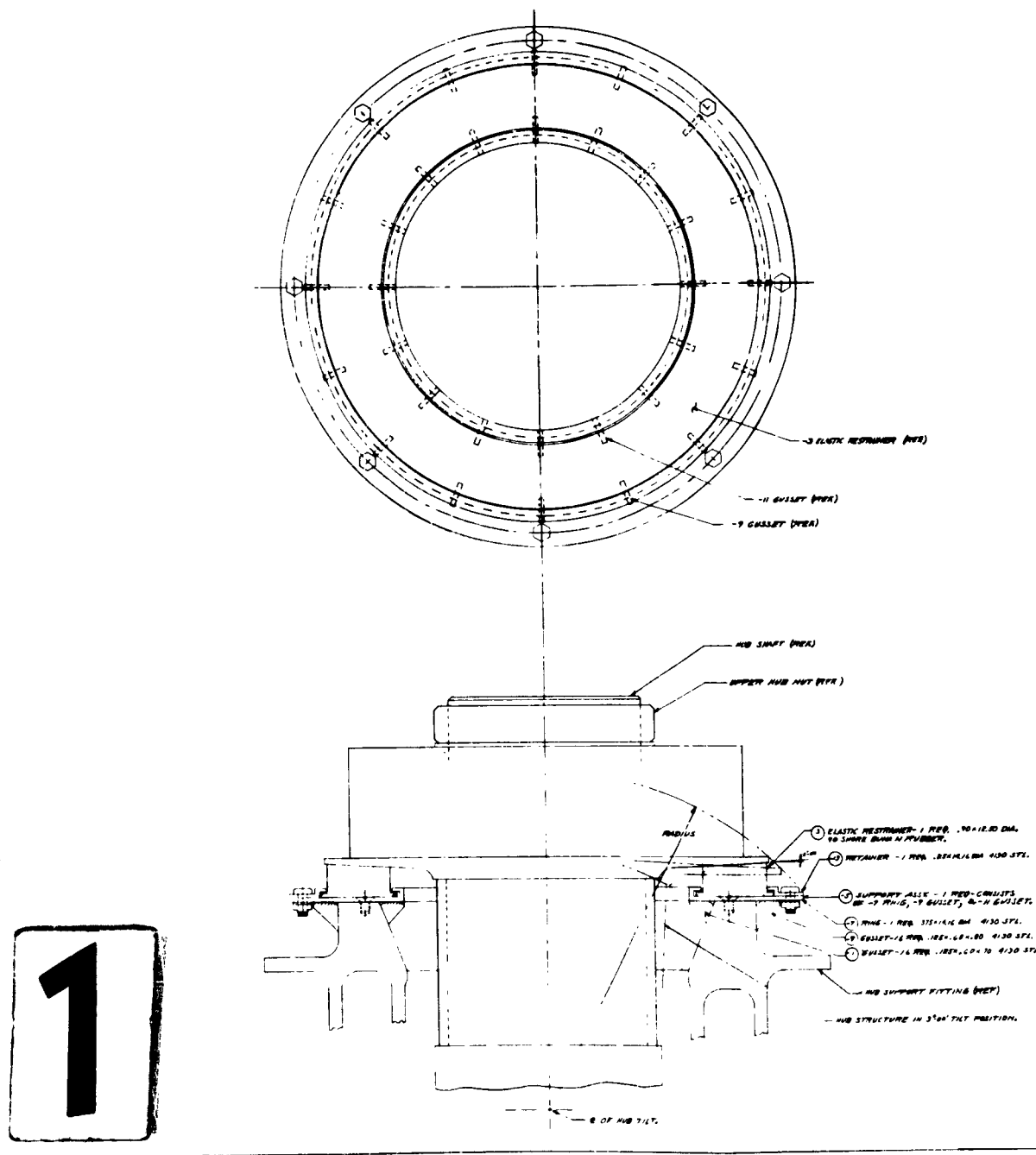
Figure 19. Elastic Restraint Installation-Hot Cycle Helicopter Hub

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-3 ELASTIC RESTRAINER (REF)

-11 GUSSET (REF)

-9 GUSSET (REF)

HUB SHAFT (REF)

UPPER HUB NUT (REF)

ELASTIC RESTRAINER- 1 REQ. .70 × 12.50 DIA.
40 SHORE DURO N RUBBER.

RETAINER - 1 REQ. .25 × 11.50 DIA 4130 STL.

SUPPORT ASSY - 1 REQ - CONSISTS
OF -7 RING, -9 GUSSET, & -11 GUSSET.

RING - 1 REQ. .375 × 11.16 DIA 4130 STL.

GUSSET -16 REQ. .125 × .68 × .90 4130 STL.

GUSSET -16 REQ. .125 × .60 × 70 4130 STL.

HUB SUPPORT FITTING (REF)

HUB STRUCTURE IN 3°00' TILT POSITION.

Helicopter Hub.

REQ'D	PART NO	REQ'D	PART NO	NAME	SIZE	DESCRIPTION	SPECIFICATION	
	ASSEMBLY OPP.		ASSEMBLY REQ'D			LIST OF MATERIAL		
					DRWN	ELASTIC RESTRAINT		
					CHK'D	INSTAL- HOT CYCLE		
					APP'D	HELICOPTER HUB	385-1202	
					APP'D			
NEXT ASS'Y USED ON					APP'D			
APPLICATION	QTY REQD				SCALE FULL			

VIEW C-C
(OF DIVERTER SUPPORT)

SECTION A-A
(AT FIREWALL)

PARTIAL VIEW SHOWN
NORMAL TO PLANE OF
DIVERTER VALVE SIDE
OUTLET

AIRCRAFT

NOTE: MAIN ACCESS PANELS
ON REAR SIDE SHOWN
REMOVED IN THIS VIEW

Figure 20. Detail Layout-Propulsion System.

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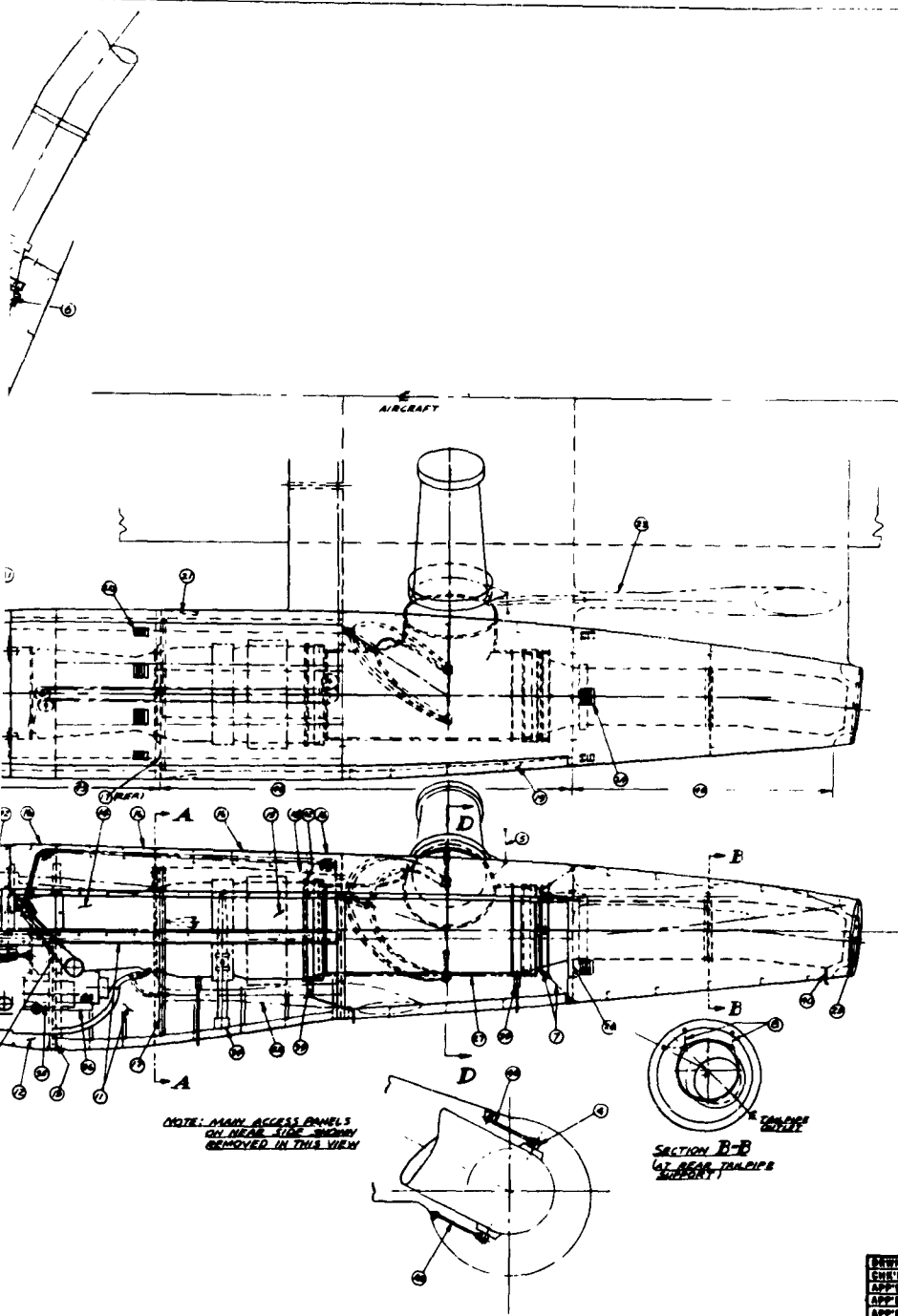
NOTES:

1. ENGINE MOUNT - FWD MAIN
   THRUST, VERTICAL, & SIDE LOADS
2. ENGINE MOUNT - REAR
   VERTICAL & SIDE LOADS
3. ENGINE MOUNT STABILIZER LINK
   VERTICAL LOADS ONLY
4. DIVERTER VALVE YOKE
   ABSORBS LOADS FROM DIVERTER INTERNAL
   PRESSURES
5. DIVERTER WEIGHT OVERHANG STABILIZER LINK
6. BALL JOINTS, EACH END OF DIVERTER, TORQUE-FREE,
   PRESSURE SEALED, QUICK DISCONNECT ATTACH.
7. FORWARD TAILPIPE SUPPORT GIMBAL
8. REAR TAILPIPE SUPPORT
9. CONTINUOUS LONGERON (ENGINE THRUST LOAD
   ALIGNS WITH DIVERTER THRUST LOAD)
10. NOSE FAIRING & INLET DUCT ATTACH TO ENGINE
    WITH 'V' BAND CLAMP. ATTACHED QUICK REMOVAL
    ACCESS PANELS TO HAVE SEMI-FLOATING ATTACHMENTS
11. HINGED ENGINE REMOVAL & ACCESS PANELS
12. QUICK REMOVAL ACCESS PANELS. FASTENER
    DESIGN TO PERMIT STORAGE ON HINGED
    ACCESS PANELS
13. THIS FRAME IS ATTACHED TO HINGED PANELS
    AND INTERLOCKED AT SPLIT JOINTS TO PROVIDE
    CONTINUITY EFFECT
14. QUICK REMOVAL ACCESS PANEL TO BE CAPABLE
    OF ABSORBING PART OF ENGINE MOUNT SYSTEM
    TORSION FOR OVERALL STIFFNESS
15. REMOVABLE ACCESS PANEL MAY BE ATTACHED
    WITH QUICK REMOVAL LOAD CARRYING SCREWS
    OR QUICK ACTING FASTENERS PENDING STRESS,
    DEFLECTION, AND WEIGHT STUDIES
16. LOAD CARRYING LOCAL ACCESS PANELS AS REQUIRED
17. SEALED FIREWALL
18. SEALED TUNNEL FOR ENGINE CONTROLS &
    PLUMBING
19. DIVERTER ACCESS PANEL, STRESSED ATTACHED
    WITH QUICK REMOVAL LOAD CARRYING ATTACHMENTS
20. ENGINE ACCESSORY COMPARTMENT COOLING LOUVRES
    FINAL NUMBER & LOCATION SUBJECT TO FURTHER
    STUDY
21. ENGINE BURNER & TURBINE COMPARTMENT COOLING
    AIR INLETS FINAL LOCATION SUBJECT TO FURTHER STUDY
22. POTENTIAL ACCESSORY COMPARTMENT COOLING AIR
    EJECTOR, POWERED BY ENGINE COMPRESSOR BLEED
    AIR DISCHARGES OVERBOARD. TO BE USED ONLY
    IF TESTS INDICATE NECESSITY
23. POTENTIAL "HOT SECTION" COOLING AIR EJECTOR;
    POWERED BY ENGINE TURBINE GAS DISCHARGES
    UNDER SIDE OF STUB WINGS. IS SHUT-OFF WHEN
    TURBINE GAS IS DIVERTED AFT. COOLING IS THEN
    ACHIEVED BY TAILPIPE EJECTOR.
24. TAILPIPE COOLING LOUVRES
25. TAILPIPE COOLING AIR EJECTOR ACTS ONLY WHEN
    TURBINE GAS IS DIVERTED AFT
26. AIR BAFFLE TO BE USED IF ITEM 6 IS USED. PRO-
    VIDE SIMPLE FLAPPER VALVES
27. DIVERTER VALVE BLANKET (REQ. BY ENG. MANUFACTURER)
28. ENGINE POWER CONTROL & LOAD SIGNAL DRIVE CABLES
29. ROTOR SPEED SIGNAL TO ENGINE CONTROL GOVERNOR;
    SYNCHRONOUS MOTOR (HYD OR ELECT) IS GEAR BOX.
    FLEXIBLE DRIVE SHAFT TO ENGINE CONNECTION
30. OIL TANK APPROX 5 GAL GROSS CAPACITY, 3 GAL OIL CAP
    PRESSURIZED VENT SYSTEM (5 P.S.I.G. MAX.)
    AIR SEPARATOR BAFFLE
    CLEANABLE SUMP WITH OIL OUTLET STANDPIPE
31. OIL FILLER CAP & STICK GAGE
32. HYDRAULIC PUMP
33. TACHOMETER GENERATOR
34. AIR IMPINGEMENT STARTING NOZZLE (MIL STD)
35. PROVISIONS FOR ENGINE MOUNTED GENERATOR OR
    OTHER ACCESSORY ON ENGINE STARTER DRIVE PAD
36. OIL COOLER - ENGINE FUEL-OIL HEAT EXCHANGER
37. TURBINE GAS DUCTS TO ROTOR.
    INCORPORATE SEALED BALL JOINTS
    2 PLACES, EXPANSION JOINT 1 PLACE (TORQUE FREE)
38. DIVERTER VALVE OUTLET REDUCER. SIZED TO
    ESTABLISH DESIRED RESULTANT LOAD DIRECTION
    FOR DIVERTER YOKE 4
39. PRESSURE ACTUATED DRAIN VALVES (MODIFIED
    ENGINE PART)
40. FIXED ORIFICE DRAIN
41. CHECK VALVE
42. "V" BAND CLAMPS FOR QUICK ENGINE CHANGE
43. FIRE ZONE 'A' FIRE DETECTORS AND CF₃Br
    FIRE EXTINGUISHING SYSTEM
44. FIRE ZONE 'B' OVERHEAT & FIRE DETECTORS.
    CF₃Br FIRE EXTINGUISHING SYSTEM
45. ZONE 'C' OVERHEAT & FIRE DETECTORS ONLY.
    RECEIVES PART OF EXTINGUISHING AGENT FROM
    ZONE 'B'
46. AUXILIARY SUPPORT STRUTS

AIRCRAFT

NOTE: MAIN ACCESS PANELS
ON NEAR SIDE SHOWN
REMOVED IN THIS VIEW

SECTION B-B
(AT REAR TAILPIPE
SUPPORT)

SECTION D-D

DRAWN		
CHK'D		
APP'D		
APP'D		
APP'D		
APP'D		
APP'D		
APP'D		

DETAIL LAYOUT -
PROPULSION
SYSTEM

HUGHES TOOL COMPANY
AIRCRAFT DIVISION

385-7002

SCALE 1/10

2

Figure 21. Detail Layout-Engine and Diverter Valve Installation.

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RING SPLICED FOR REMOVAL

DIAGONAL STIFFENER
OR MEMBER

VIEW AT A-A

VIEW AT B-B

VIEW AT C-C

TYPICAL AT
THREE MAIN
RINGS (MINI-
MUM)

OIL TANK VENT
OUTLET
(SEE FRONT VIEW)

DRAIN
HOLES

DOWEL SHEAR PINS
& TENSION LATCHES
OR TIES

2

FENER

VIEW AT B-B

VIEW AT C-C

VIEW AT D-D

ADJUSTABLE LINKS

VIEW AT E-E

SECTION L-L

SECTION F-F

DOWEL SHEAR PINS
& TENSION LATCHES
OR TIES

RAIN
OLD
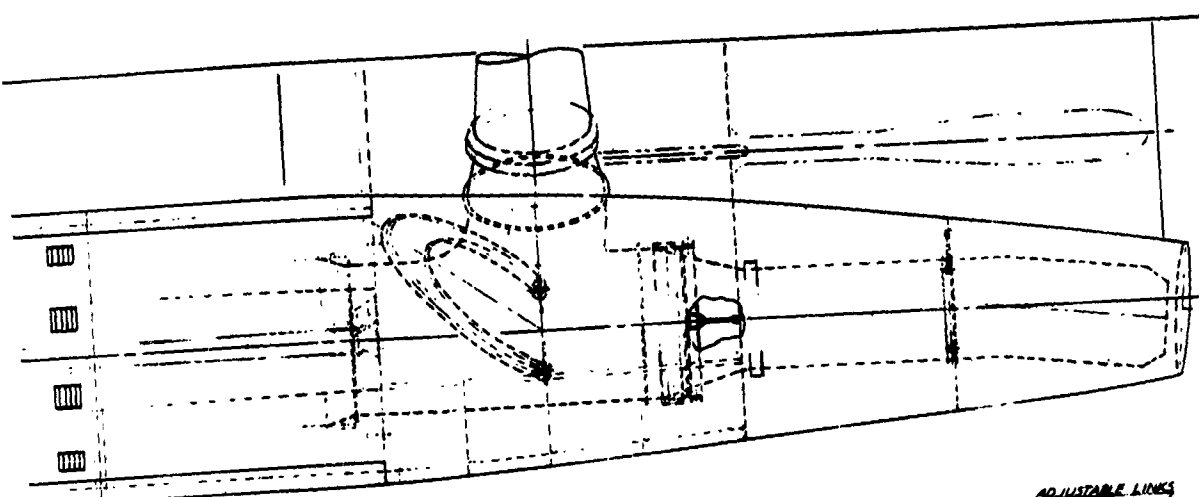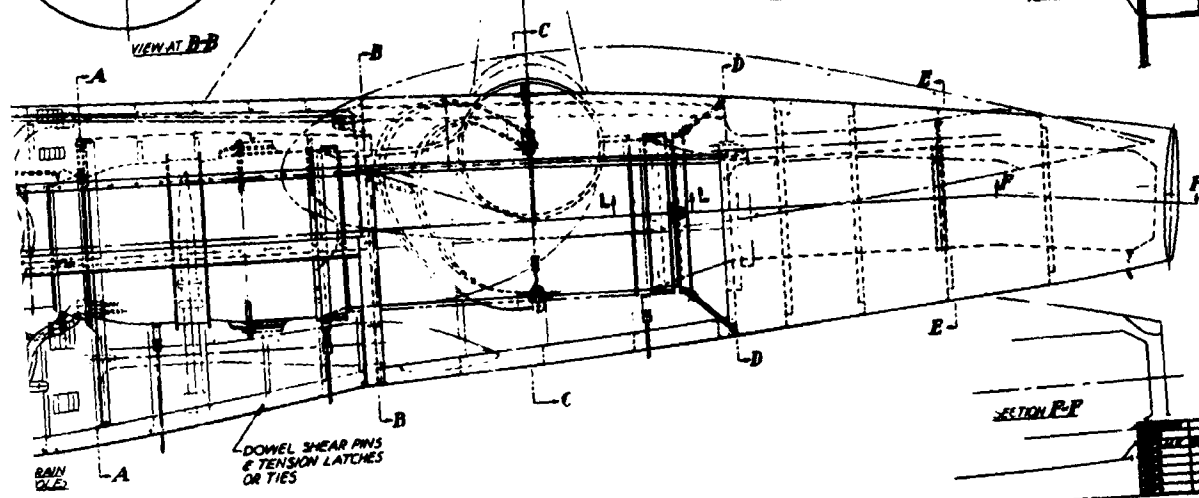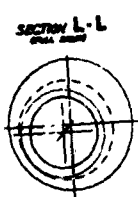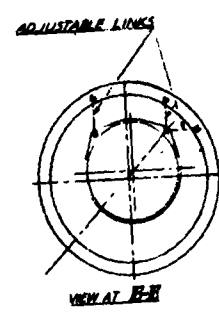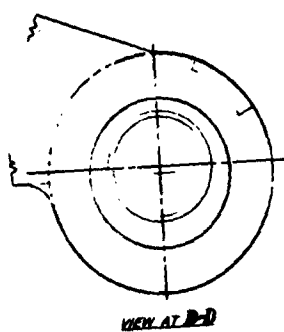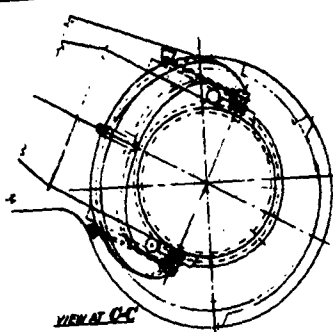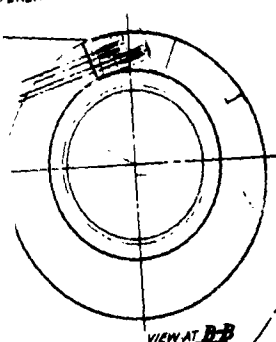
DETAIL LAYOUT—
ENGINE &
DIVERTER VALVE
INSTL

3

setting in proportion to rotor speed variation. The load signal presets the fuel flow setting to minimize the effect of engine loading upon transient and steady state power turbine speed. The research aircraft gas generators have no power turbine. Instead, the rotor speed signal (at a nominal 243 rpm) substitutes for the power turbine signal which normally transmits into the fuel control at 4,995 rpm when power turbine speed is 17,000 rpm. Therefore, a speed multiplier is required. This speed multiplier is provided in the accessory gear box which is described below. Obviously, this rotor speed signal will require essentially zero power from the accessory gear box, but the signal must be produced at the proper rpm to make the fuel control function properly. The rotor speed output shaft from the accessory gear box will drive a hydraulic pump which will be connected in a 1:1 speed relationship to a hydraulic motor mounted directly in front of the speed signal pad on the fuel control. A short flexible shaft will connect the hydraulic motor and the fuel control at the speed signal pad.

A cable and pulley system was selected to operate the two-engine control shafts for the following reasons:

a. The power control shaft has a 120-degree nominal travel and the load signal shaft has a 120-degree range including overtravel. Poor linearity and rigging conditions would occur with these travels, if a rod and ball crank system were used.

b. The cable system is lighter than a push rod and bell crank system.

c. The nacelle contours require a compact control system which can fit inside the contour. Cables proved to be more adaptable to this condition.

9.2.5   Propulsion System Mounting (See Figures 13, 20, and 21)

Limitations on the YT64 aft section permissible loads required isolation of diverter valve forces from the engine casing. Consequently, the propulsion mounting system consists of three elements:

a. A conventional jet engine type of mounting system for the engine.

b. A diverter load absorption yoke.

c. A tailpipe support system.

The engine mount supports form a statically determinate system. A ball stud, fixed to engine at one inboard forward mounting pad, is retained in the socket of a tripod-type mount which is attached

to the nacelle structure in a stable manner. This support point reacts vertical, side, and thrust loads. Another spherical bearing is mounted in a fitting attached to the nacelle structure just aft of the engine center of gravity. An inverted V-link system connects this spherical bearing to the rear upper engine support lugs. A crowned spacer (integral with a link) is utilized to provide a rolling action on the bore of the engine mount lugs corresponding to the action of the spherical bearing on the structure. This support point reacts vertical and side loads only, with no fore and aft restraint. Engine expansion is consequently permitted by this support point.

The remaining engine support is a ball rod-end stabilizer link located on a forward engine mount pad opposite the thrust reaction point. This support reacts in a vertical direction only. Moments in a plane normal to the engine longitudinal axis are reacted by the two forward mount members.

The diverter valve produces large forces as the result of the action of engine gas pressure on the surfaces of the diverter shell and valves. Since flexible joints are required at all three ends of the diverter to prevent transmission of excessive thrust or radial loads to the engine, rotor ducts, and tailpipe, these large-magnitude pressure forces must be reacted by a primary load-carrying system which will, however, permit relative motion of these components due to deflection of the airframe and nacelle structure. Comparison of several proposed parallel L and V strut systems and their attachment to the structure indicated that for the number of degrees of freedom required by the system, a yoke-type support was superior in action and simplicity. The extra weight of the yoke is offset by the additional support attachment fittings and multiple load path structural reinforcement of the other proposed systems.

The diverter valve side outlet reducing transition duct is sized so the resultant of the diverter valve forces acts in the plane of the yoke when the engine gas is diverted 90 degrees. When the gas discharges straight through the diverter valve, the forces are small and result only from the pressure loss through the valve and small changes in momentum. For the one-engine-out condition and intermediate valve positions, the component of load out of the yoke plane is reacted by two auxiliary support struts.

The weight of the diverter is supported by the engine aft end and a stabilizing link at the side outlet transition duct.

All connections to the diverter valve are sealed spherical-type joints with expansion provisions. These joints resist shear, but not moments or axial forces.

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The forward tailpipe support consists of two struts and a gimbal ring assembly which carries tailpipe thrust loads to the aft main nacelle frame. This system is designed to support part of the tailpipe weight by the diverter valve, but excludes diverter loads from being transmitted into the relatively thin unstiffened walls of the tailpipe.

The tailpipe rear support consists of three struts, two vertical and one horizontal. This support reacts side load and vertical load, but does not resist fore-aft loads, and therefore permits tailpipe expansion. The links of the aft support are adjustable to permit tailpipe alignment.

9.2.6   Nacelle Structure

The front and rear spars of the nacelle pylons extend as full circular frames around the diverter and tailpipe. The depths of these frames are such that they may be easily modified in the future to provide structure for an outer wing panel. In order to provide fire resistance to basic structure, the caps and stiffeners of these main frames will be stainless steel. The webs will be either titanium or stainless steel. The latter will permit spotwelding to the caps and stiffeners and is therefore preferred. The thin web does not involve a large weight increment, and stainless steel maintains structural integrity in case of fire.

The nacelle skin surrounding the diverter valve between the main frames is titanium with stiffener rings of the same material. The longerons are stainless steel. The diverter access door is aluminum alloy skin with titanium stiffeners.

Aft of the rear main frame, the nacelle skin and stiffeners are aluminum alloy except for a short section adjacent to the tailpipe cooling air ejector. This section is titanium. The ejector is fabricated from stainless steel.

The upper 120-degree segment of the nacelle, forward of the front main nacelle frame, constitutes the primary engine support structure. It is cantilevered from the main frames. The continuous longerons on each side of the section are stainless steel as are the frames at the engine support fittings. The skin and frames forward of the sealed stainless steel firewall are aluminum alloy, except as previously stated. A stainless steel stringer is used on the top centerline between the firewall frame and the front engine mount frame. Aft of the firewall, the skins and frame are titanium.

Access panel support longerons, near the nacelle horizontal centerline, are continuous from a disconnect fitting on the engine inlet nose piece to the front main nacelle ring at the pylon spar station.

These longerons carry the hinge for the engine removal doors and help support air loads. The access panels between the upper nacelle segment and the access panel support longeron assist the upper segment in carrying nacelle torsion loads from engine mount reactions. The forward access panel is aluminum alloy and the aft panel is titanium. The access panel support longerons are titanium.

The use of the materials outlined above is considered to provide sufficient support to carry the weight of a nonoperating engine in event that fire should damage or burn away portions of the aluminum alloy structure.

Hinged engine removal and access panels on the lower portion of the engine nacelle are latched and doweled with shear pins at the bottom split line. The upper ends of the forward, rear, and intermediate main frames of hinged access panel engage pins, anchored to the upper structure frame stations, to take radial loads. The hinged access panels, skins, frames, and stiffeners are aluminum alloy material.

Access panels between the fixed nose piece and the hinged doors on the lower half of the nacelle and between the nose piece and the forward structural frame on the upper half of the nacelle are aluminum alloy. They are attached by quick-acting fasteners spaced at relatively close intervals around the periphery of the nose piece to preclude any tendency of the panels to "lift" from air loads. The panels incorporate stiffener rings immediately aft of this joint for the same reason. Fasteners at the aft end of the panels incorporate floating action in tangential directions to provide relief of nacelle deflections relative to the engine inlet attachment to which the nose piece is clamped.

The nose fairing and engine inlet duct are fabricated as a unit of aluminum alloy. Air loads in the axial direction are transmitted basically to the engine inlet flange with some assistance provided by the access panel support longerons on the horizontal center line.

The nose fairing shape will be designed to provide efficient engine inlet pressure recovery in both hovering and forward flight as well as good aerodynamic external contours for forward flight at the higher speeds.

9.2.7   Nacelle Compartment Cooling

Cooling of the engine accessory compartment forward of the fire wall, the engine hot section zone, the diverter valve zone, and the tailpipe compartment are related to fire prevention considerations.

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The air inlets and outlets must be strategically located for cooling air circulation. In addition, these openings should be located to minimize potential hazards in the event of fuel or oil line failures in the accessory compartments.

The preliminary arrangement for accessory compartment cooling calls for a series of louvres around the periphery of the nacelle immediately forward of the firewall. A gap in the series is provided just above the horizontal centerline to clear flush ram scoop-type inlets for the engine hot section compartment air inlet.

For the present, no other air openings are proposed for the accessory compartment. If tests show that additional inlets are needed in the forward areas, they will be located after a "tuft" study of the engine inlet flow field to prevent entry of inflammable fluids into the engine compressor in the event of a line failure.

Location of the engine hot section cooling air inlets near the horizontal center line just aft of the firewall appears to be a reasonable compromise between air circulation needs and minimization of inflammable fuel entry possibility.

The diverter valve is covered with an insulation blanket. This is required by the manufacturer to suppress temperature gradients of the structural portions of the diverter valve assembly. Because of the heat insulating effect of the diverter blanket, the preliminary design does not incorporate air openings in the nacelle surrounding the diverter.

Three sets of louvres, spaced 120-degrees apart, are proposed in the tailpipe area just aft of the rear main nacelle frame. When the tailpipe is exhausting the engine gases during ground run, an ejector at the outlet draws air through these louvres and the engine hot section air inlets forward. During forward flight, ram air supplements the ejector action.

Air circulation over the engine hot section during hover conditions is provided by an engine exhaust gas powered ejector which draws air through this compartment. This ejector operates only when the engine gases are diverted to the rotor ducts. Automatic switchover to the tailpipe ejector system occurs when the gas is diverted aft. The hot section ejector is located in the lateral pylon as close to the nacelle as possible. Its application is tentative pending ground rig test of circulation characteristics of air through other openings such as at the tailpipe outlet, and trial louvres. If the ejector is used, it is anticipated that an air baffle may become necessary to block reverse flow through the tailpipe ejector. Simple flapper check valves would have to be incorporated in the baffle to permit air flow through the tailpipe ejector in one direction.

If added cooling should be required for the engine accessory compartment, an auxiliary ejector operated by engine compressor bleed is proposed. The compressor air should be cooled by expansion to a temperature not hazardous in the event of inflammable fluid line failures. The ejector would be exhausted overboard to prevent entry of such fluid into the engine hot section.

9.2.8   Oil Tank and Cooling System

A schematic drawing of the engine oil cooling system is shown in Figure 22. An aluminum alloy oil tank is supported on the nose assembly. Air separation from the oil is achieved by baffling and impinging the mixture on the outer concave surface of the tank wall. Centrifugal action forces the heavy oil against the wall to release the air. A pressurizing valve with a fixed orifice bleed is used to suppress foaming. Normally, this valve is set at 4 psi to maintain engine oil pump inlet pressures at 5 psi minimum absolute for high-altitude operation. Since the research aircraft ceiling is approximately 20,000 feet, the fundamental purpose of the pressurizing valve will be to suppress foaming tendencies and limit the vent outlet orifice automatically for all conditions. The gross oil tank volume of 5 gallons provides for 40 percent airspace above a 3-gallon filling level. The large air space assists in deaeration and prevention of discharge of oil overboard.

Ground tests will be conducted to demonstrate compliance with the engine manufacturer's maximum permissible aeration of 10 percent by volume.

Separate vent connections are utilized for the tank vent and the engine sump vents to minimize overboard discharge of oil vapors from the engine sumps.

The oil tank sump is provided with an opening large enough to permit hand access for cleaning.

A standard oil filler cap and a 10-mesh filler screen are provided on the tank filler neck. An oil quantity stick gauge is also provided.

Oil cooling is accomplished by means of an engine fuel to oil cooler. The cooling unit plumbing arrangement is to be either supplied or recommended by the engine manufacturer.

9.2.9   Engine Starting

Air impingement starting is utilized. The ground connection will be a standard ground air start coupling, accessible through a door on the under side of each nacelle. Crossmanifolding is not presently
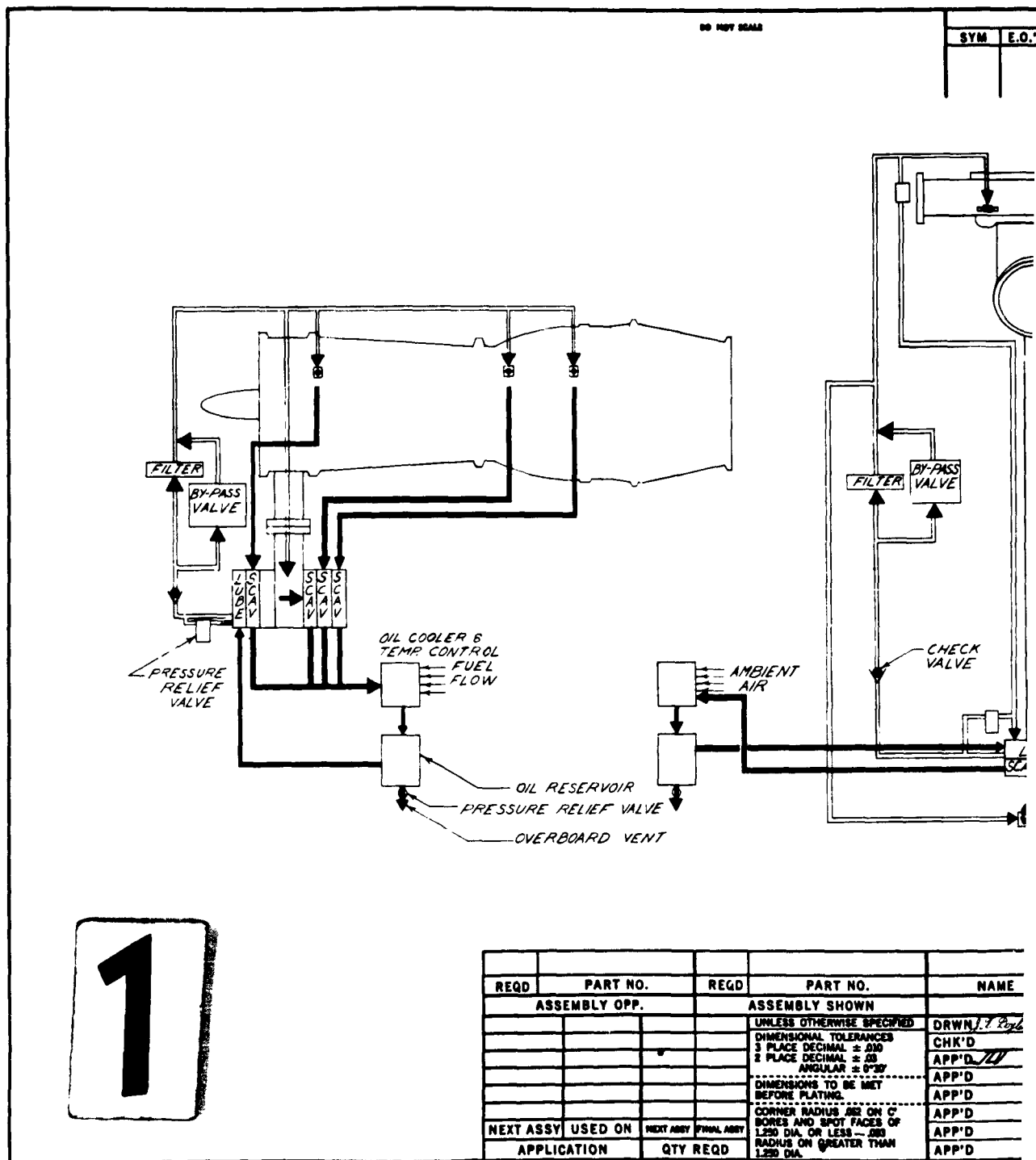
Figure 22. Engine and Rotor Lubrication Systems - Schematic.

DO NOT SCALE

ROTOR

FILTER

BY-PASS VALVE

CHECK VALVE

LUBE SCAVENGE

9305

OIL COOLER & TEMP CONTROL

FUEL FLOW

AMBIENT AIR

OIL RESERVOIR

PRESSURE RELIEF VALVE

OVERBOARD VENT

▬▬ SCAVENGE & LOW PRESSURE OIL
═══ HIGH PRESSURE OIL

REQD	PART NO.	REQD	PART NO.	NAME	SIZE	DESCRIPTION	SPECIFICATION
ASSEMBLY OPP.		ASSEMBLY SHOWN		LIST OF MATERIAL			

UNLESS OTHERWISE SPECIFIED
DIMENSIONAL TOLERANCES
3 PLACE DECIMAL ± .010
2 PLACE DECIMAL ± .03
ANGULAR ± 0°30'
DIMENSIONS TO BE MET BEFORE PLATING.
CORNER RADIUS .002 ON C' BORES AND SPOT FACES OF 1.250 DIA. OR LESS — .003
RADIUS ON GREATER THAN 1.250 DIA.

| DRWN | CHK'D | APP'D | APP'D | APP'D | APP'D | APP'D | APP'D |

ENGINE AND ROTOR
LUBRICATION SYSTEMS -
SCHEMATIC

HUGHES TOOL COMPANY
AIRCRAFT DIVISION
CULVER CITY, CALIFORNIA

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NEXT ASSY | USED ON | NEXT ASSY | FINAL ASSY
APPLICATION | QTY REQD

SCALE

rication Systems - Schematic.

5

2

contemplated but may be incorporated at a later date so that compressor bleed air from one engine may be used to start the opposite engine.

9.2.10 Engine Accessories

Basic accessories mounted on the engine are:

a. Hydraulic Pump

b. Gas Generator Tachometer

c. Rotor Speed Servo Motor

d. DC Generator.

9.2.11 Plumbing

A sealed tunnel running from the firewall to the forward main nacelle frame at the front spar is proposed to separate fuel, hydraulic, fire extinguisher, and other plumbing from the hot section of the engine. The engine control cables are also routed through this tunnel.

In addition to the engine-equipped fuel manifold and combustion chamber pressure-closed drain valve, two similar types of valves are installed at the low point of the diverter adapter ducts. Their purpose is to permit drainage of raw fuel from these low points in case of false starts. A fixed orifice drain tube is provided at the tailpipe nozzle for the same purpose.

9.2.12 Engine, Diverter Valve, and Tailpipe Removal

Engine removal is accomplished through the hinged lower access panels. The separate access panels forward of the hinged doors may be stored on the hinged panels by the rear fasteners of the panels. The engine is lowered after detachment either on an engine dolly or by means of a cable winch from above. A cable attachment access hole will be provided in the top of the nacelle. The nose piece with the attached oil tank is removed with the engine.

Quick-detachable, V-band clamps are used to attach the air inlet duct and the diverter adapter section to the engine exhaust flange. A bolted fitting is provided where the nose piece attaches to the access panel support longerons. This fitting will be designed to permit the nose piece to clear the longeron when installing or removing the engine.

The diverter valve and the tailpipe are removed through the large access door incorporated in the outboard lower portion of the nacelle between the main frames. This access door is stressed in flight and is attached with quick-removal screws of the multiple-thread, high-lead type (or similar type).

Flexible seals at the diverter ends are of a quick-detachable design. The tailpipe is moved aft through the main rear frame inner diameter enough to clear the diverter by disconnecting the tailpipe supports. The diverter is disconnected from the yoke and auxiliary struts by removal of trunnion spindle retention bolts and lowered from the nacelle.

9.2.13   Fire Detection and Extinguishing System

The fire detection and overheat system is shown schematically in Figure 23 and is a continuous-element, repeatable-type, operating through transistorized control boxes. The accessory compartment system forward of the firewall forms one system loop while the engine hot section including the diverter valve and tailpipe forms another loop. Each loop is a continuous nonredundant circuit to permit a full continuity ground check.

The accessory compartment forward of the firewall is treated separately because it is fundamentally a different type of zone, carrying inflammable fluids and a more likely source of potential fire. The hot section is more likely to be the source of overheat indication from failed hot gas joints or diverter valve failure. Separation of the systems gives the pilot a clue as to the type of emergency. Also the location of a faulty sensing element is made less difficult during continuity checks. The proposed detection system gives detailed consideration to overheat detection in the area of the diverter attachments.

Control boxes are available which differentiate between fire warning and overheat. The separate loops of the system detect both through separate warning lights. Fire detection in the tailpipe area was not considered mandatory, but the warning system is simpler with than it is without such indication. Separate control boxes are used for each loop since these units are readily available, whereas dual units are special.

The fire extinguishing system selected for this application is a single-shot, double-capacity CF_3B_r system, discharging into the engine accessory compartment, the engine hot section, and the diverter valve zones simultaneously. Distribution of the agent is through tubing perforated with 0.049 inch minimum diameter holes. The double quantity is discharged proportionately into the zones in less than 4 seconds. This is at the same rate per pound of agent as required for a basic quantity of agent, i.e., less than 2 seconds.

CONTAINER FOR
CF$_3$Br FIRE EXTINGUISHING.
NITROGEN PRESSURIZED

WIRING TO
OVERHEAT &
FIRE WARNING
INDICATOR
LIGHTS.

HINGED ACCESS
DOOR (REF.)

OIL TANK

FIREWALL
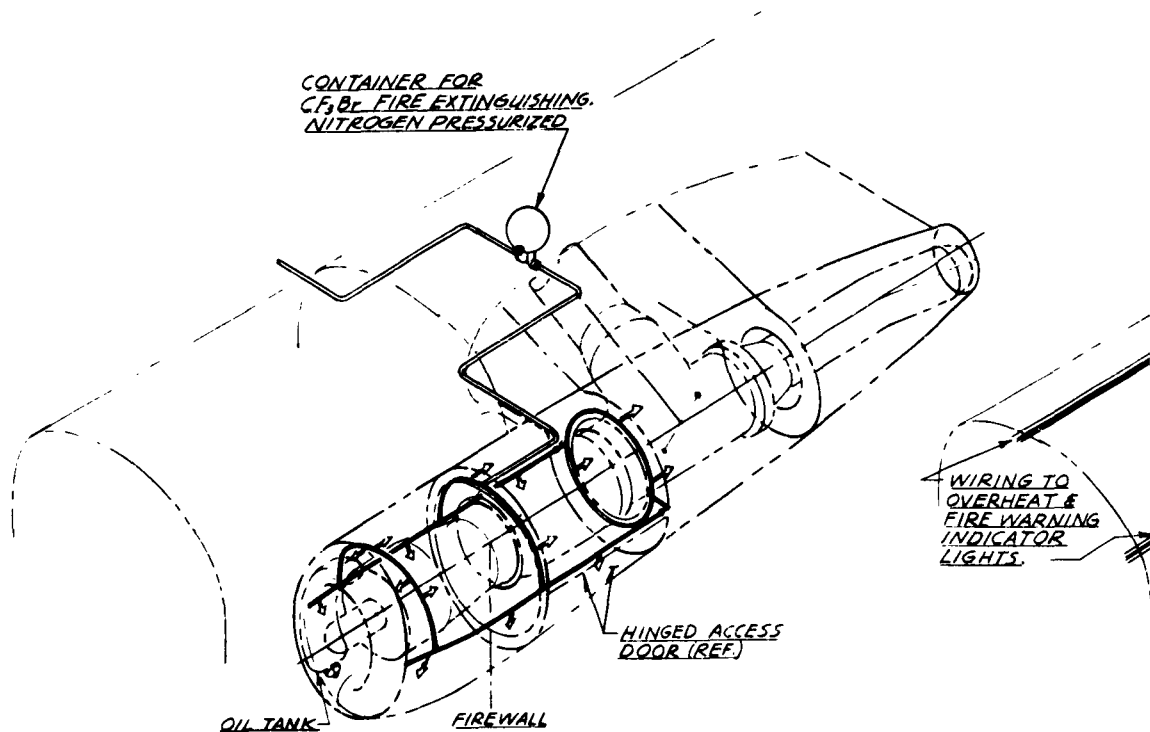
FIRE EXTINGUISHING SYSTEM
NOTE: ARROWS INDICATE DIRECTION
OF DISCHARGE OF PERFORATED
TUBES.

PRELIMINAR
HOT CYCLE HELI
AND EXTING

Figure 23.   Fire Detection and Extinguisher System - Schematic.

TRANSISTORIZED
CONTROL BOXES

CONTINUOUS ELEMENT
TYPE DETECTOR

TO CONTROL BOX

ACCESS PANELS (REF)

FIREWALL
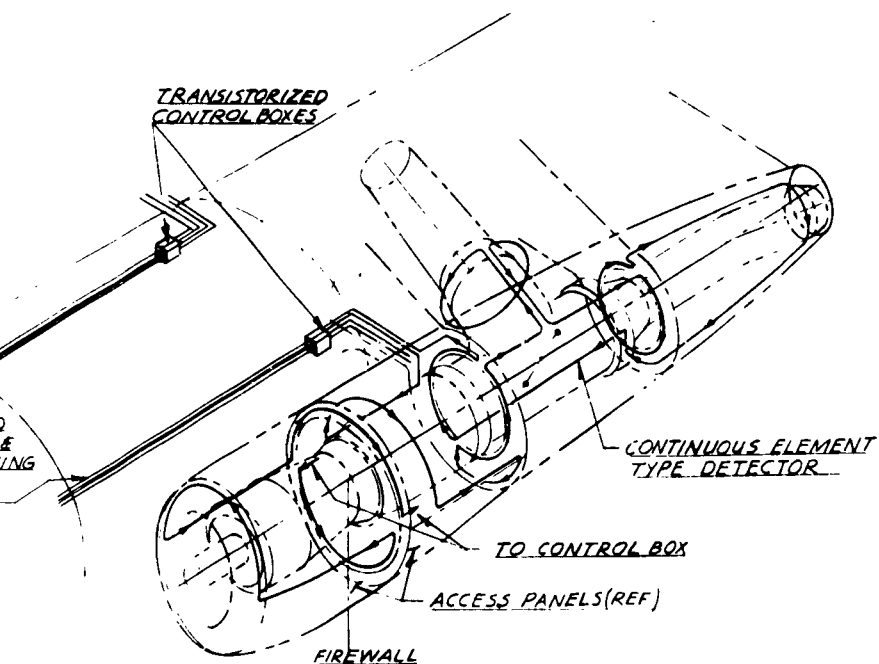
FIRE AND OVERHEAT DETECTION SYSTEM

NOTE: ARROWS INDICATE SYSTEM CONTINUITY

INARY SCHEMATIC
HELICOPTER FIRE DETECTION
XTINGUISHING SYSTEM

DRWN W.W.K		SCHEMATIC - FIRE DETECTION & EXTINGUISHER SYSTEM	HUGHES TOOL COMPANY AIRCRAFT DIVISION CULVER CITY, CALIFORNIA
CHK'D			
APP'D			
APP'D			
APP'D			385 -8203
APP'D			
APP'D			
APP'D		SCALE NONE	CODE 02731 SHEET OF

A high-rate discharge system (H. R. D.) was not used because of the extensive testing and development involved in obtaining the precise distribution and concentrations required by this type of system under all flight conditions. For an experimental aircraft of the type involved, the possible cost and time delays of such a system were considered not warranted.

The zone characteristics used to establish the quantity of agent required are as follows:

Zone Designation	Net Volume (cu. ft.)	Assumed Max. Cooling Air Flow (Fwd. Flight) (lb/sec)
"A" (engine access. compt.) fwd. of F. W.	15.5	0.25
"B" (from F. W. to rear main nacelle frame and to stub wing barrier)	26	0.4

The basic quantity of agent required was determined by use of the formula from Spec. MIL-E-5272(USAF)

Agent Quantity (LBS) $= (0.56 \, W_a + 0.16 \, V)$

Wa = Air flow through zone, pounds per second

V = Net volume of zone, cubic feet

This quantity was then doubled to compromise between the quantity required for production aircraft and that required by the USAF for experimental aircraft.

The proposed system will permit basic bottle charge pressures, and line and orifice sizes while sustaining the basic rate and concentration for twice the basic length of time. This should give added insurance that zone shape and air flow path irregularities will be adequately handled and will also provide margin for increase in cooling air flow as the result of testing.

The zone surrounding the tailpipe was not included in determining the amount of agent required since it is not considered subject to hazardous quantities of inflammable fluids. A portion of the agent

from the diverter and engine hot section zones will flow through this zone when the agent is discharged.

The total quantity of agent is established at 14 pounds. It is carried in a single container located on the centerline of the aircraft. The container is equipped with two electrically activated, dual squib operated, frangible disc-type outlet fittings, one for each engine system, and a thermal relief plug. The resulting system is simple in action and has a minimum number of components. Switches in the pilot compartment simply select one engine or the other to receive the entire charge. All fire extinguishing lines in the nacelles are 0.028-inch wall stainless steel. The lines in the nacelle pylon and fuselage outside potential fire damage areas are 0.035-inch wall aluminum alloy. Steel fittings are used in and adjacent to the nacelles and aluminum alloy in the fuselage and pylon.

Fire detection elements and fire extinguishing lines were not routed in the lower portion of the nacelle to avoid attaching such equipment to the engine. Attachment of these elements to the hinged doors and access panels was held to be undesirable. The fire detector element is, however, routed in a complete circle around the firewall station periphery. Special attention is given to the upper portion of the oil tank above the oil level in respect to both detection and fire extinguishing. Fire in the lower portion of the nacelle should be quickly detected by the elements on the lower side of the access panel longerons or on the firewall near the cooling louvres. The fire extinguishing lines routed on the lower side of the same longerons discharge agent towards the bottom of the nacelle.

9.2.14   Fuel System

The fuel system is shown on Figures 14 and 24. It is of typical aircraft design outlined in HIAD Vol. I, Part D. Two fuel tanks, one 260 gallon and one 240 gallon, are individually serviced with the filler openings in the fuselage. Under normal operation, each engine receives fuel from its own tank. However, for emergency operation, fuel is available to both engines from either tank. Fuel is supplied to the engines through 3/4-inch diameter lines. Included in the system are the required standard valves, strainers, vents, drains, and pumps.

9.2.15   Rotor Driven Accessory Gear Box

A small rotor-driven accessory gear box will be required to drive the following items:

a. Hydraulic pump to actuate the flight controls; approximately 15 horsepower.

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Figure 24. Fuel System - Schematic Diagram.

FUEL TANK VENT

SYPHON BREAKER
VALVE

FUEL TANK

B.P.

1¼ DIA x .035
ALUM TUBING

¾ DIA x .035
ALUM. TUBING

¾ DIA x .020
CRES TUBING
THRU ENG. COMPT.

-12 HOSE

ENG #2
(REF)

STRAINER
DRAIN

BOOSTER
PUMP DRAIN

FIREWALL (REF)

B.P.

FUEL TANK

SUMP DRAIN

-7220
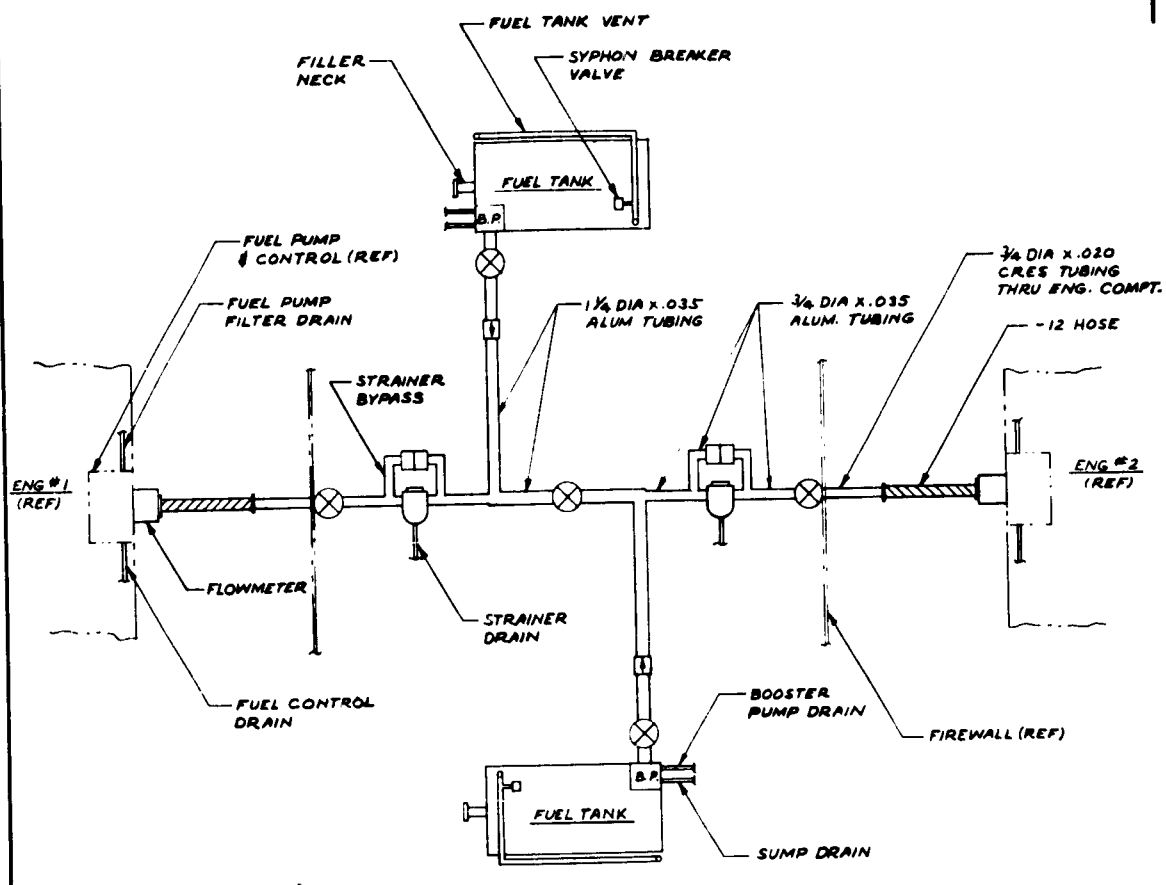
$\boxed{\downarrow}$ — CHECK VALVE

$\bigcup$ — FUEL STRAINER

$\boxed{\square}$ — BYPASS VALVE

$\boxed{B.P.}$ — BOOSTER PUMP

$\bigotimes$ — SHUT-OFF VALVE

EQUIP. SYMBOLS

REQD	PART NO.	REQD	PART NO.	NAME		SIZE	DESCRIPTION	SPECIFICATION
ASSEMBLY OPP.			ASSEMBLY SHOWN			LIST OF MATERIAL		
			UNLESS OTHERWISE SPECIFIED	DRWN FARSHOR	5/25/60			
			DIMENSIONAL TOLERANCES 3 PLACE DECIMAL ± .010 2 PLACE DECIMAL ± .03 ANGULAR ± 0°30'	CHK'D				
				APP'D LT	6/14/60			
				APP'D				
			DIMENSIONS TO BE MET BEFORE PLATING.	APP'D				
			CORNER RADIUS .082 ON C' BORES AND SPOT FACES OF	APP'D				
NEXT ASSY	USED ON	NEXT ASSY	1.250 DIA. OR LESS — .093	APP'D				
APPLICATION		QTY REQ'D	RADIUS ON GREATER THAN 1.250 DIA.	APP'D		SCALE NONE		

FUEL SYSTEM-
SCHEMATIC DIAGRAM

HUGHES TOOL COMPANY
AIRCRAFT DIVISION
CULVER CITY, CALIFORNIA

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Schematic Diagram.

2

b. Hydraulic pressure pump to lubricate the rotor hub bearings; approximately 1 horsepower.

c. Hydraulic scavenge pump on return line from hub bearings; approximately 1 horsepower.

d. Rotor speed tachometer.

e. Hydraulic pump used to provide rotor speed signal (equivalent power turbine speed signal) to T64 gas generator fuel control.

As shown on Figure 14 this small gear box will be located at the lower end of the mast and will be driven by a gear fastened directly on the mast. The gear box will include the rotor thrust bearings and the five power take-off pads noted above. All components will be lubricated by an integral circulating oil system.

9.2.16 Tip Cascade Valve

In case of failure of one engine, the rotor blade tip nozzle area would have twice the area required to keep one engine on its proper operating line. The power output with such a large nozzle area would be less than half that which would be available if the tip nozzle area were closed down to the proper value. A design of a two-position nozzle system, called the tip cascade valve, is given in Figure 25. This valve system consists of two properly placed rotating doors, which are normally in line with the gas flow, and offers minimum restriction when both engines are operating. When an engine failure is indicated, the two doors are automatically rotated by means of cables, bell cranks, gear sectors, and connecting links to a deflected position which will block off approximately 50 percent of the nozzle area associated with the fore and aft ducts of each rotor blade. Adequate seals are built into the periphery of the movable doors to prevent gas leaks in the closed position.

This design and operation of the tip cascade valve is discussed more fully in Reference 24. Detail drawings of the components of the valve system shown in Figure 25 are given here in Figures 26, 27, 28, 29 and 30 for information only.

9.3 AIRFRAME

The general configuration of the airframe of the research vehicle is shown in Figure 14. This drawing shows the frame, stringer, and skin construction at several typical sections along the fuselage. A more complete discussion of the type of construction chosen and a brief analysis of basic loads is given in Section 7. However, sufficient detail

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is included on Figure 14 to show that the fuselage structure is conventional. The frame spacing, stringer spacing, and floor location, for instance, all represent standard practice which further analysis will substantiate.

9.4 FLIGHT CONTROLS

The flight controls of the research aircraft are designed to permit flight both as a helicopter and later as a high-speed autogyro. The helicopter type of longitudinal, lateral, and collective pitch control is conventional, but the yaw control is obtained with reaction jets supplied by gas bled from the engines. The autogyro control involves an additional coordinated motion of the vee-tail in the "elevator" regime of the conventional horizontal tail and this feature will be utilized toward the end of the test program.

9.4.1 Helicopter Flight

The flight control system is shown in Figures 31 and 32. Figure 31 shows the control system from the cockpit to the swashplate, including HO-6 helicopter control sticks in the cockpit, HO-6 mixer assembly, push rods, and bell cranks to the lower stationary swashplate. Figure 32 shows the rotor controls from the swashplate to the rotor blades. Figure 31 also shows the push rod and cable system that operates the jet yaw control in hovering and helicopter flight, and the interconnection to the vee-tail to cause it to function as a rudder whenever the rudder pedals are operated. The pedals are always connected to the vee-tail so it will always be operated with the pedals, even in hovering when it is ineffective.

9.4.2 Jet Yaw Control

As shown in Figures 33 and 34, engine gases are bled from the rotor supply ducts in hovering and forward flight as a helicopter through suitable ducting to two variable area nozzles in the aft portion of the fuselage. The nozzle exit opening for each valve is controlled by a pressure-balanced flapper-type valve. The nozzles direct gases to the left or right to produce the desired yawing moment in helicopter flight. The valves are attached by cables to the directional control pedals in the cockpit and are closed in the neutral position. Since the gas pressure tends to open the basic valve, the balance chamber is sized to overcome this tendency, and, also to provide an adequate rudder pedal force gradient.

The variation of nozzle area is obtained by progressively exposing or closing some of the five equal area exits in each nozzle.
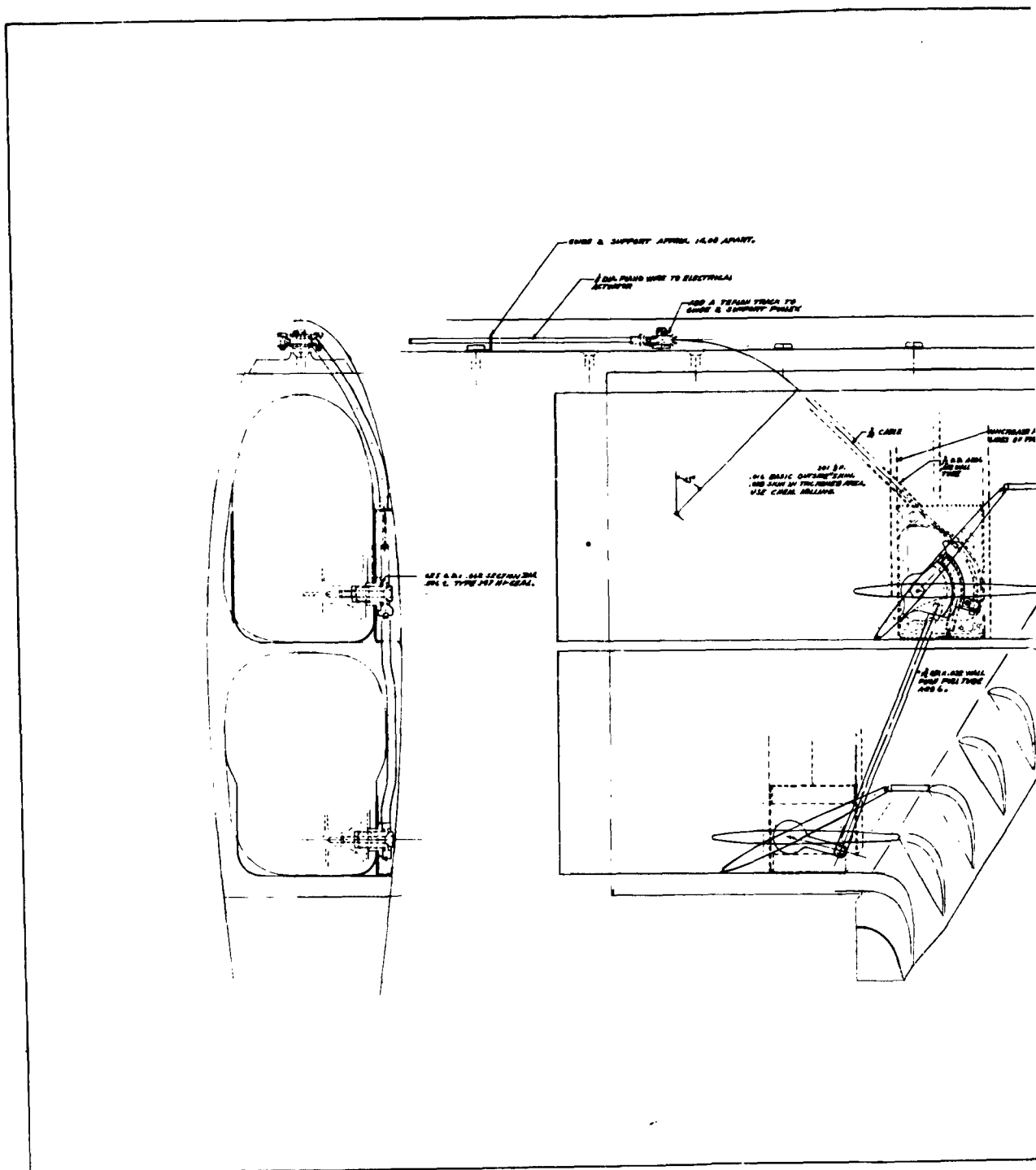
Figure 25.   Layout - Cascade Valve.

117

STA. 333.18

2
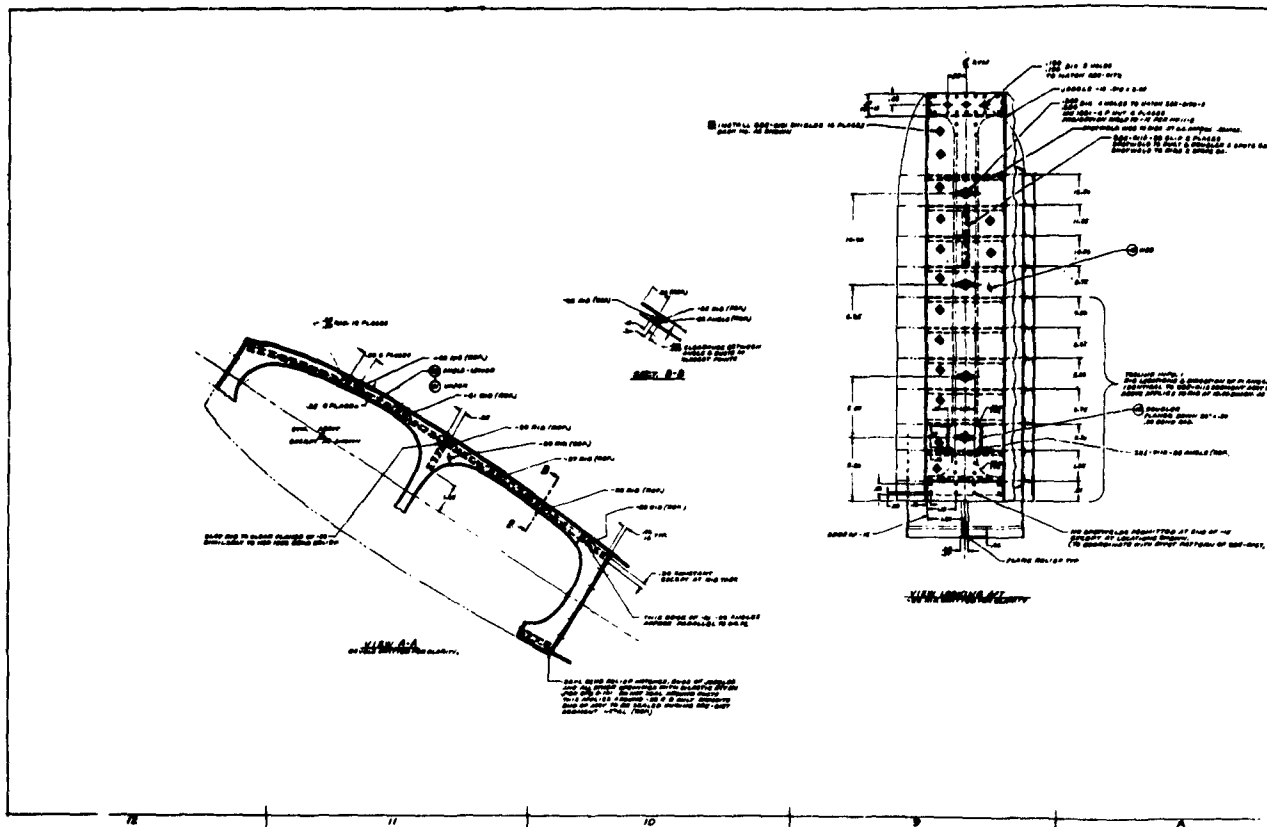
Figure 26.   Tip Assembly - Forward, Sheet 1.

119
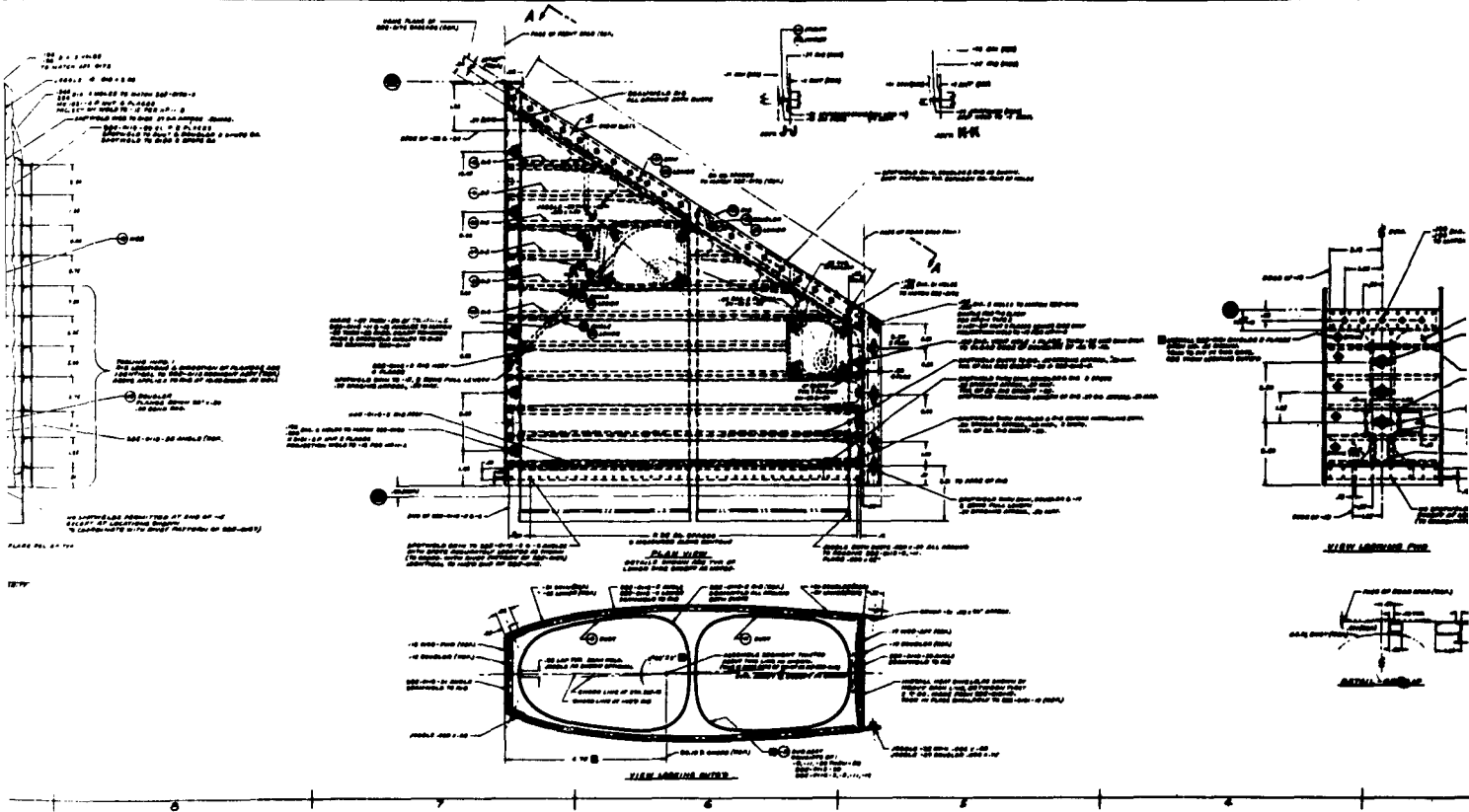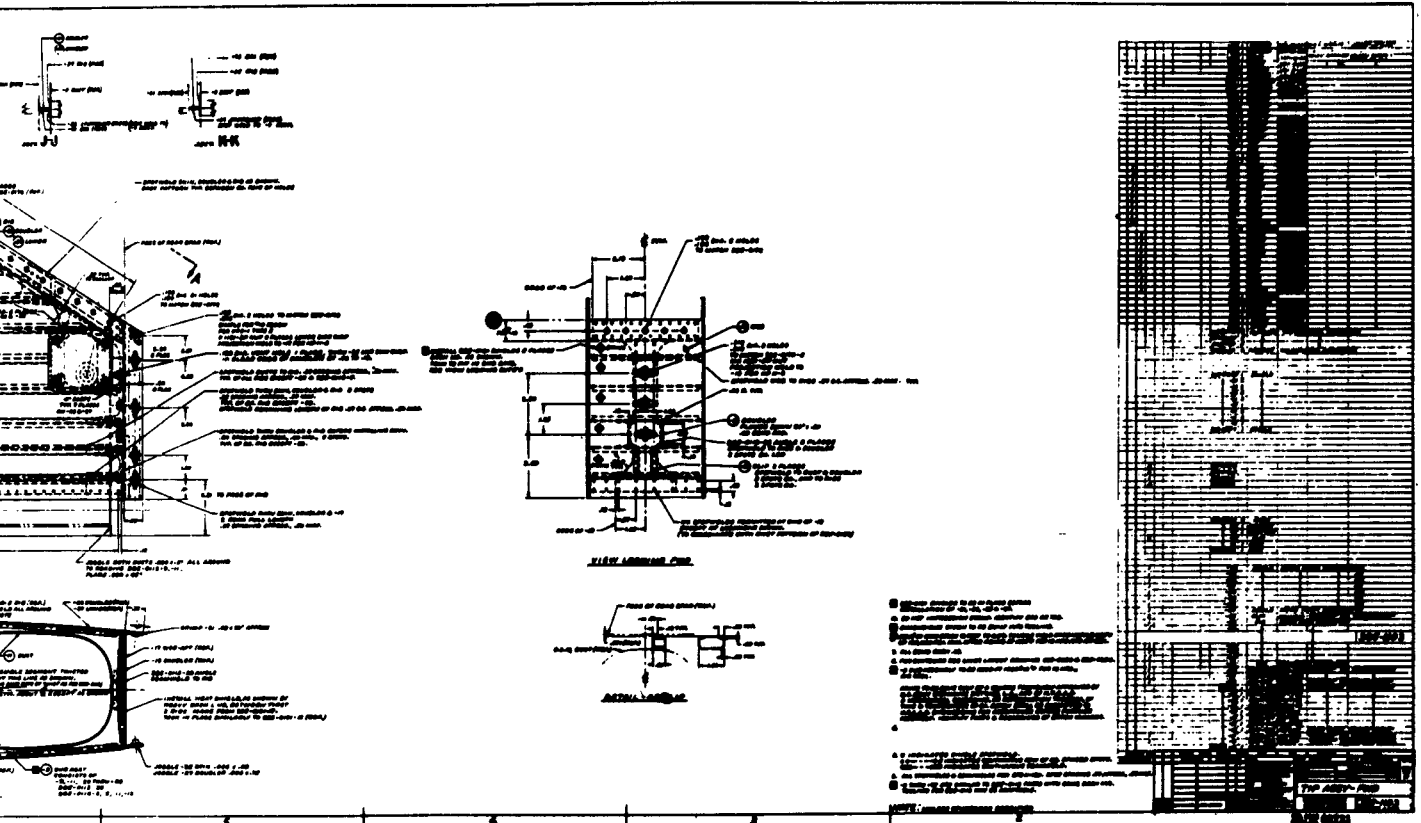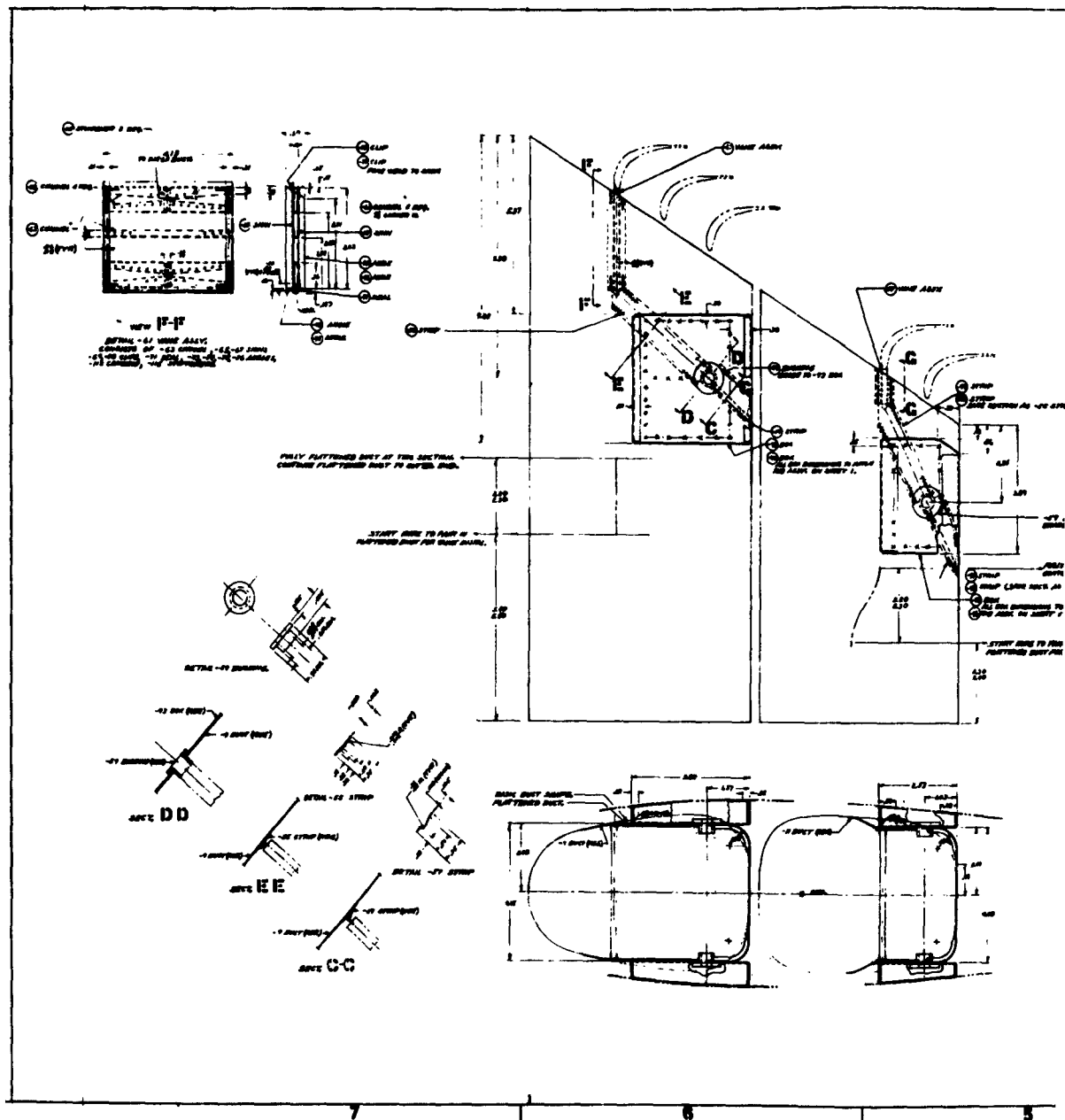
Figure 26. Tip Assembly - Forward, Sheet 2.

121

VIEW G-G

H

H

2

TIP ASSY- FWD

3

Figure 27. Cascade Valve - Forward Duct.

CIRCULAR SPACE WIDTH
MIN EFFECTIVE .0491
MAX. EFFECTIVE .0503 (REF.)
MIN. ACTUAL .0504 (REF.)
MAX. ACTUAL .0516

.0540
DIA. PIN

C A
D B

INTERNAL SPLINE DATA

FLAT ROOT SIDE FIT
N/A OF TEETH 14
PITCH 32/64
PRESSURE ANGLE 30°
A MAJOR DIA. .4600 - .4718
B MINOR DIA. .4081 - .4111
C PITCH DIA. .4375
D MEASUREMENT BETWEEN PINS MAX .3634 (NO.)
TIR ALL SPLINE SURFACES

SPLINE SPACES ALONG THIS F.

5.65

(-9) SEAL
(-7) SEAL

13

.18 R.
-21

(-3) FRAME
-5 SKIN (REF.)

A
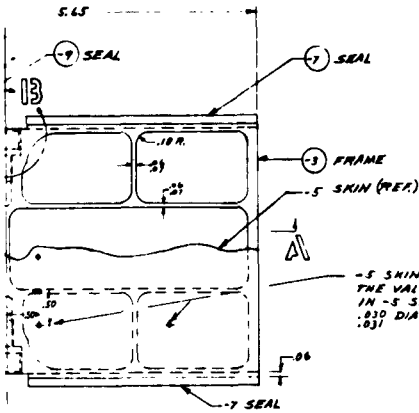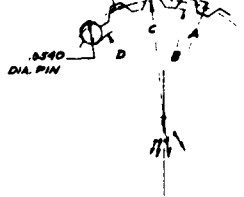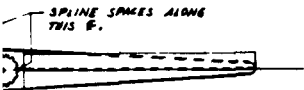
-5 SKIN TO BE THE AFT SKIN WHEN
THE VALVE IS IN CLOSED POSITION.
IN -5 SKIN ONLY DRILL OR PUNCH ONE
.030 DIA. HOLE IN EACH BAY AS SHOWN.
.031

.50

.06

-7 SEAL

.092

.160

AFTER BRAZING SKIN IN PLACE
ROUND TRAILING EDGE TO FULL RADIUS

.050

.30 R.

A-A

385-1104

L MACHINED SURFACES.

SKINS & -7 & -9 SEALS TO -3 FRAME WITH
SILICON- BORON BRAZING MATERIAL.
BRAZING; AGE THE ASSY. AT 1900°F. -16 HRS, AIR COOL.

ASE -3 FRAME MATERIAL IN THE
ON TREATED & AGED CONDITION.

				UNLESS OTHERWISE SPECIFIED	DRWN SALLOWS 4-26-6	CASCADE VALVE - FORWARD DUCT.	HUGHES TOOL COMPANY AIRCRAFT DIVISION CULVER CITY, CALIFORNIA
				DIMENSIONAL TOLERANCES	CHK'D		
				3 PLACE DECIMAL ± .010	APP'D		
				2 PLACE DECIMAL ± .03	APP'D		
				ANGULAR ± 0°30'	APP'D		
				DIMENSIONS TO BE MET BEFORE PLATING	APP'D		
385-1100	385-1000	1	3	CORNER RADIUS ARE ON 'C' BORES AND SPOT FACES OF .125 DIA. OR LESS — .010 RADIUS ON GREATER THAN 1.25 DIA.	APP'D		385-1104
NEXT ASSY	USED ON	NEXT ASSY FINAL ASSY			APP'D		
APPLICATION		QTY REQD			APP'D	SCALE FULL	CODE 02731 SHEET OF

2

Figure 28.   Lever - Aft Duct Cascade Valve.

DO NOT SCALE

CIRCULAR SPACE WIDTH
MAX. EFFECTIVE .0491
MIN. EFFECTIVE .0479 (REF)
MAX. ACTUAL .0478 (REF)
MIN. ACTUAL .0466

.0600
DIA. PIN

EXTERNAL SPLINE DATA	
FLAT ROOT SIDE FIT	
NO. OF TEETH	14
PITCH	32/64
PRESSURE ANGLE	30°
A MAJOR DIA.	.463 - .461
B MINOR DIA.	.3960-.3880
C PITCH DIA.	.4375
D MEASUREMENT OVER PINS.	MIN. .5379 (REF)
RD ALL SPLINE SURFACES	

385-1105

.120 DIA. THRU
90°X .31 DIA. C'SINK.

.525 Q.D.

.63 DIA.

.050
.094

.070

.175

.194

2.00 R.

30° CHAMFER

EXTERNAL SPLINE (SEE DATA CHART)

.156
.313

.50 R.

1.500
1.656
2.126

.177 DIA. THRU.

2

REQD	PART NO.	REQD	PART NO.	NAME	SIZE	DESCRIPTION	SPECIFICATION
		1	-3	LEVER	1.00 X 1.50 X 2.00	RENE 41 BAR SOLUTION TREATED & AGED	GENERAL ELECTRIC B50T59-52
	ASSEMBLY OPP.		ASSEMBLY SHOWN			LIST OF MATERIAL	

	UNLESS OTHERWISE SPECIFIED		
DRWN SALLONG			
CHK'D		LEVER - AFT	HUGHES TOOL COMPANY
APP'D		DUCT CASCADE VALVE	AIRCRAFT DIVISION CULVER CITY, CALIFORNIA
APP'D			
APP'D			
APP'D			
APP'D			385-1105

385-1100	385-1000	2	6
NEXT ASSY	USED ON	NEXT ASSY	FINAL ASSY
APPLICATION		QTY REQD	

SCALE

CODE 02731 SHEET OF

Duct Cascade Valve.

Figure 29. Sector - Forward Duct Cascade Valve.

127

CIRCULAR TOOTH THICKNESS
MAX. EFFECTIVE .0491
MIN. EFFECTIVE .0479 (REF)
MAX. ACTUAL .0478 (REF)
MIN. ACTUAL .0466

.0600
DIA. PIN

EXTERNAL SPLINE DATA	
FLAT ROOT SIDE FIT	
NO. OF TEETH	14
PITCH	32/64
PRESSURE ANGLE	30°
A MAJOR DIA.	.463 - .461
B MINOR DIA.	.3960 - .3880
C PITCH DIA.	.4375
D MEASUREMENT OVER PINS	MIN. .5379 (REF)
TRI ALL SPLINE SURFACES	

.075 R.
.150
BREAK SHARP EDGES.

(-5) TUBE
(-7) SECTOR
FULL WELD
BOTH SIDES.
(-3) SECTOR

GROOVE .625 O.D.

(TYP) SLOT WIDTH

.094

2.00 R.
.250 DIA. .260 DP
.251
.062 CORNER R.
.200
.400

3B5-1106

ON ALL MACHINE SURFACES.

1	-7	1	-7	SECTOR	1.00 x 1.50 x 3.00	WAFE 41 BAR SOLUTION TREATED & AGED.	GENERAL ELECTRIC B 50T59-S8	
1	-5	1	-5	TUBE	.375 O.D. x 1.0 STK WALL x 2.50 LONG	WAFE 41 TUBE ANNEALED	"	
1	-4	1	-3	SECTOR	1.00 x 2.00 x 2.00	WAFE 41 BAR SOLUTION TREATED & AGED	"	
REQD	PART NO.	REQD	PART NO.	NAME	SIZE	DESCRIPTION	SPECIFICATION	
-2 ASSEMBLY OPP.		-1 ASSEMBLY SHOWN			LIST OF MATERIAL			

UNLESS OTHERWISE SPECIFIED
DIMENSIONAL TOLERANCES
2 PLACE DECIMAL ± .010
3 PLACE DECIMAL ± .005
ANGULAR ± 0°30'
DIMENSIONS TO BE MET
BEFORE PLATING.
CORNER RADIUS .015 ON C
BORES AND SPOT FACES OF
1.250 DIA. OR LESS — .030
RADIUS ON GREATER THAN
1.250 DIA.

DRWN	SALTOW	
CHK'D		
APP'D		
APP'D		
APP'D		
APP'D		
APP'D		

SECTOR - FORWARD
DUCT CASCADE VALVE.

3B5-1106

3B5-1106	3B5-1000			
NEXT ASSY	USED ON	NEXT ASSY	FINAL ASSY	
APPLICATION		QTY REQD		

SCALE FULL

CODE 02731 SHEET OF

2

Figure 30. Cascade Valve - Aft Duct.

129

CIRCULAR SPACE WIDTH
DIM. EFFECTIVE .0491
MAX. EFFECTIVE .0503 (REF)
MIN. ACTUAL .0504 (REF)
MAX. ACTUAL .0516

.0540
DIA. PIN

D C A
B

—SPLINE SPACES ALONG
THIS ℄.

—-7 SEAL (REF)

INTERNAL SPLINE DATA	
FLAT ROOT SIDE FIT	
NO. OF TEETH	14
PITCH	32/64
PRESSURE ANGLE	30°
A MAJOR DIA.	.468-.4718
B MINOR DIA.	.4001-.4111
C PITCH DIA.	.4375
D MEASUREMENT BETWEEN PINS MAX.	34.3400REF
TYP ALL SPLINE SURFACES.	

5.42

(-7) SEAL

(-7) SEAL

(-3) FRAME

.190
18 PLACES

-.24

—-5 SKIN REF. (-5 SKIN TO BE THE AFT SKIN WHEN
THE VALVE IS IN CLOSED POSITION.
IN -5 SKIN ONLY DRILL OR PUNCH ONE
.250 DIA. HOLE IN EACH BAY AS SHOWN.

.06

-.1 SEAL

-.160

.180

—AFTER BRAZING SKIN IN PLACE,
ROUND TRAILING EDGE TO FULL RADIUS

2

385-1107

…NED SURFACES.

…INS 2, -7 & -9 SEALS TO -3 FRAME WITH
…ON-BORON BRAZING MATERIAL.
…NG; AGE THE ASS'Y AT 1400°F-16 HRS., AIR COOL.

…FRAME MATERIAL IN THE
…ATED & AGED CONDITION.

	REQD	PART NO.	REQD	PART NO.	NAME	SIZE	DESCRIPTION	SPECIFICATION
	1	-11			SKIN	.010×4.50×6.80	NENE 41 SPRM ANNEALED	GENERAL ELECTRIC REQ'D-52
	1	-9			SEAL	.006×3.00×5.80	NENE 41 SPRM ANNEALED	GENERAL ELECTRIC REQ'D-52
	2	-7			SEAL	.006×.60×3.70	" "	" " "
	1	-5			SKIN	.010×4.50×6.00	NENE 41 SPRM ANNEALED	GENERAL ELECTRIC REQ'D-52
	1	-3			FRAME	.150×4.50×6.00	NENE 41 PLATE	" " "
REQD	PART NO.		REQD	PART NO.	NAME	SIZE	DESCRIPTION	SPECIFICATION
ASSEMBLY OPP.			ASSEMBLY SHOWN				LIST OF MATERIAL	

UNLESS OTHERWISE SPECIFIED
DIMENSIONAL TOLERANCES
2 PLACE DECIMAL ± .03
3 PLACE DECIMAL ± .010
ANGULAR ± 0°30'
DIMENSIONS TO BE MET
BEFORE PLATING.
CORNER RADIUS .015 OR C-
FORNER AND SPOT FACES OF
.035 DIA. OR LESS = .005
RADIUS ON GREATER THAN
.035 DIA.

DRWN	SALLOWS	12-1-62
CHK'D		
APP'D		
APP'D		
APP'D	JLV	Trake
APP'D		
APP'D		
APP'D		

CASCADE VALVE —
AFT DUCT

HUGHES TOOL COMPANY
AIRCRAFT DIVISION
CULVER CITY, CALIFORNIA

385-1100	385-1000	1	3
NEXT ASSY	USED ON	NEXT ASSY	FINAL ASSY
APPLICATION		QTY REQD	

385-1107

SCALE FULL		CODE 02731	SHEET OF

Figure 31. Layout - Hot Cycle Primary Control System.

131

3

Figure 32. Layout - Hot Cycle Rotor Control System.

133

Tongue Tube 1

4/2 Shoes

Tongue Tilt

VL212 74

Tongue Jo ass.

Upper Roter Bearing

Lower Stationing Drip

Lower Roter & Thrust Bearing.

VL.22 16

Cruise Level Flight (Control Stick Neutral)

Fenthering Arm

Hin Shoft

Lower Roter 10

2/4 Collective

15.30

Collar

Figure 33.   Layout - Yaw Control Duct Installation.

135

2

3
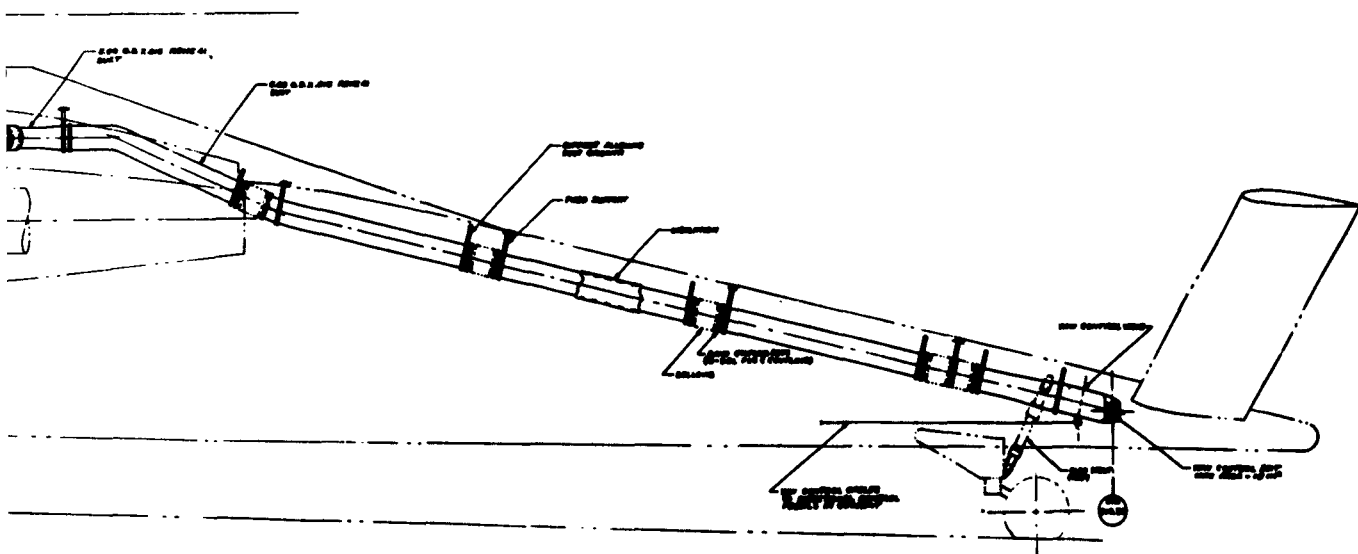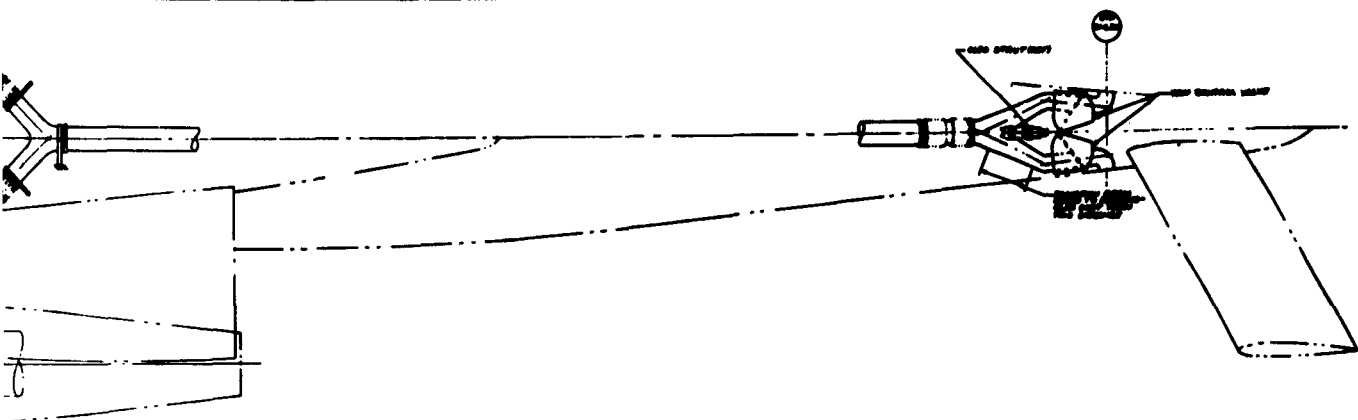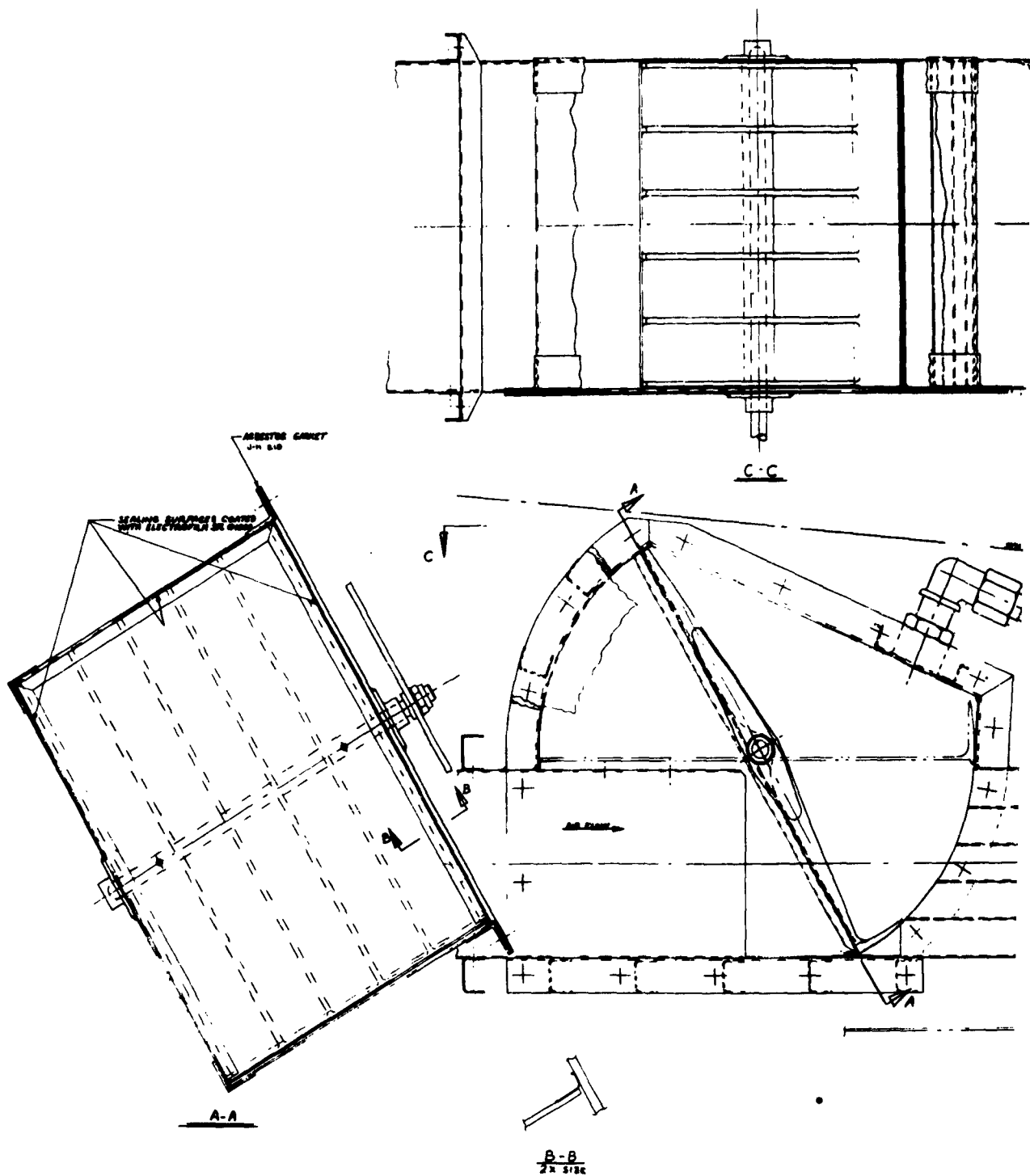
Figure 34. Layout - Yaw Control Valve.

137

VENT TO ATMOS.

MAX. EXIT AREA - 10 IN²
3 IN² PER ORIFICE

STR
614.00

C

2

EXIT AREA - 16 IN²
PER CASCADE

$\downarrow_c$

LAYOUT–
YAW CONTROL
VALVE

385-4350

3

The gases bled from the rotor supply ducts pass to the nozzles through Rene' 41 ducting, which is insulated to protect the surrounding structure. Suitable bellows are installed in this ducting to allow for thermal expansion.

9.4.3 Autogyro Flight

For autogyro flight the rotor controls operate in the same manner as in helicopter flight, with the controls shown in Figures 31 and 32. However, the diverter valves will be in the through-flow position to send gases through the tail pipe and rearward facing nozzles to obtain forward thrust from autogyro flight. With the diverter valves in this position, no gases are available to the yaw control jets which tap off gases from the rotor supply ducts. Consequently, the jet yaw controls will not be effective. Nevertheless, the directional pedals will cause the movable portions of the vee-tail to function as a rudder, in the same fashion they operated in helicopter flight. In the autogyro regime, the rudder action will be the primary yaw control, instead of acting as an auxiliary to the jets as in helicopter flight.

In the initial flight test of the research aircraft, longitudinal control in autogyro flight will be obtained with longitudinal cyclic control only. After the initial flights, and to secure additional longitudinal control power, the vee-tail will be interconnected to the stick to produce elevator-type motion which will aid in obtaining control in high-speed flight. A simple mixing linkage will be added to the vee-tail control system to permit elevator operation of the vee-tail.

9.5 LANDING GEAR

The landing gear for the research aircraft is the two main wheel-tail-wheel-type and is shown in Figures 5 and 14. The landing gear is essentially the same as that for the H-34 helicopter and has been in service use for several years. The basic geometry of the landing gear and its structural capability were obtained from Sikorsky Aircraft in References 25, 26, and 27. An examination of the strength of this landing gear when fitted to the hot cycle research aircraft is reported in Section 7.2.3 and it is seen to be adequate for the gross weights involved in test of the research aircraft.

9.6 HYDRAULIC SYSTEM

A schematic drawing of the hydraulic system is shown in Figure 35. Control system pressure of 3,000 psi is supplied by three Vickers variable displacement pumps, two of them engine driven and one rotor driven. The hydraulic pumps will feed three servo controlled actuators. Each actuator is of a tandem piston design which, when coupled to a relief valve, will allow the 3,000 psi system pressure to

be reduced to 1,500 psi acting on each piston. In the event of one engine shutdown or failure while flying, a pressure sensing valve will close the relief valve to the engine system still operating and build up the pressure to the normal 3,000 psi. Thus, the actuator will still be capable of delivering the full required load stroke performance.

At the same time, the inoperable system, again using a pressure sensing valve, will open full flow from one side of the piston to another and reduce internal drag.

In the event of a double-engine failure in flight, the rotor driven pump will feed 3,000 psi to a single system in the same manner as for a single engine failure, thus allowing the pilot full blade control during autorotation.

9.7   ELECTRICAL SYSTEM

The electrical system for the research aircraft will be conventional throughout. The generating equipment will consist of two 28-volt dc generators of 100-ampere capacity with a standard voltage regulator, reverse current, and paralleling components. A standard lead-acid battery will be used to supply the essential electrical load when the engines are not running or electric power is not otherwise being supplied to the aircraft. Standard provisions for external power will also be available.

The power utilization equipment to operate the vehicle will be minimum and will consist of standard instruments, lights, solenoids and actuators, and a radio.

Additional power utilization equipment will consist of the flight test instrumentation equipment, which will include oscillographs, temperature recorders, and photo panel equipment.

9.8   FLIGHT INSTRUMENTS

The cockpit in the HO-6 helicopter (which is the cockpit that will be used on the research aircraft) is sufficiently narrow for a single centrally located instrument panel to be provided for the pilot and co-pilot. It is felt that this same single instrument panel arrangement will be satisfactory for the research aircraft with the addition of dual indicating engine instruments, i.e., with two needles marked "left" and "right."

The proposed flight instrumentation is as follows:

Altimeter
Airspeed Indicator

Figure 35. Hot Cycle Hydraulic Control System - Schematic.

Figure 35. Hot Cycle Hydraulic Control System - Schematic.

Rotor Tachometer
Magnetic Compass
Rate of Climb Indicator
Turn and Bank Indicator
Directional Gyro
Outside Air Temperature
Fuel Quantity
Ammeter
Percent N_1 Engine Tachometer (dual)
Exhaust Gas Temperature Indicator (dual)
Fuel Pressure (dual)
Engine Oil Pressure (dual)
Engine Oil Temperature (dual)
Anti-Icing Indicator (dual)
Clock

9.9 FLIGHT TEST INSTRUMENTATION

9.9.1 Introduction

This section of the report contains a discussion of the test instrumentation to be used in the flights to be performed on the hot cycle research aircraft.

9.9.2 Purpose

The tests will be performed to obtain flight stress distribution data; blade, ducting, and structure flight temperature distribution data; and flight performance data.

9.9.3 Discussion

Instrumentation has been and will be planned to accomplish all measurements and data-gathering tasks required in the most simple, least complex manner compatible with the results required. Instrumentation will center around three major data-gathering types of equipment: recording oscillographs, a temperature recorder, and an instrument photo panel.

9.9.3.1 Recording Oscillographic Equipment. The oscillographic equipment will consist of one 50-channel oscillograph and two 18-channel oscillographs used individually or in combination, as required by the data channel needs for the specific tests being performed. Galvanometers with appropriate frequency response and sensitivities will be used as each data channel requires and will be changed as required to accommodate changing test result needs. Bridge balancing and calibrating equipment will be used, as well as remote controls. Data will be measured and recorded as follows:

a. Stress Distribution Data will be obtained by use of a standard dc-excited strain gage bridge system feeding directly through bridge balancing and calibrating equipment to galvanometers in standard recording oscillographs.

b. Engine Operation Data will be obtained by use of fuel flow tranducers, rpm tachometer generators, and pressure and temperature tranducers feeding directly into galvanometers in the oscillograph.

c. Flight Operation Data will be obtained by the use of yaw, turn and attitude rate gyros with high resolution potentiometer pick offs; airspeed and altimeter pressure transducers with high-resolution potentiometer pick offs; and spring-return, high-resolution potentiometers directly coupled for position indications of controls and other moveable items. All of these potentiometers will be part of a bridge system with the unbalanced output providing dc driving signals to galvanometers in the recording oscillographs.

9.9.3.2   Temperature Recorder - Multipoint

a. Temperatures to be recorded will include exhaust gas and duct air, cooling duct air, blade segments, spars, couplings, feathering ball, ribs, skin, shafts, hubs, etc.

b. Temperatures will be sensed by thermocouples. Outputs will feed through three 48-point switch boxes and a temperature junction box to a multipoint recorder to provide recording of 144 temperature measurements.

9.9.3.3   Photo Panel

A 35-mm movie camera with a wide-angle lens will be mounted in front of a special instrument panel containing selected flight and engine instruments. The camera and panel will be hooded to prevent stray light entry. The camera will be fix-focused on the panel and will be provided with a remote control. The camera will be operated during test flights by the pilot or test engineer. It will be coordinated with the recording equipment for identification of flight and operating condition by the counter and clock. The chief use of the photo panel records will be to provide corollary information in the event of some unusual occurrence or malfunction during flight. The film will be developed only when such conditions occur in order to make more detailed corroborative analysis. Data to be recorded by the photopanel system will include:

Altitude
Airspeed
Rate of climb

Engine and rotor rpm
Exhaust gas temperature
Engine oil temperature and pressure
Rotor oil temperature and pressure
Fuel flow
Such other indications as may be deemed desirable
Clock
Counter (synchronized with oscillograph)

9.9.3.4 Auxiliary Equipment

a. Slip rings will be provided to transmit signals from rotating components, such as blades and shafts, to the appropriate recording equipment.

b. Sound level measuring equipment will be included to determine sound levels and frequencies for analysis of possible insulation requirements.

9.10 CARGO HOOK FOR EXTERNAL LOADS

An alternate mission for the research aircraft will be demonstration of heavy-lift capability with external loading of the payload. It is planned to use the cargo hook shown in Figure 14 to conduct these heavy-lift demonstrations. This hook is to be a commercially available item and it will have four alternate methods of operation for release of load:

a. A pilot-operated pushbutton which energizes a 24-volt solenoid.

b. A manual emergency release.

c. A touchdown release which automatically releases the load upon contact with the ground.

d. A manual release operated by ground personnel.

The hook is so located that the line of action of the load will pass through or near the center of gravity of the aircraft, thus minimizing the moment due to a stationary or swinging load. A sufficiently large opening is provided in the floor to permit a 10-degree conical swing of the load in any direction without restraining the load.

The hook is attached by a rod to a universal joint which in turn is supported by a cradle attached to the rotor support structure. The hook, rod, and cradle may be removed when the aircraft is not used for heavy lift.

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ANNEX

WHIRL TOWER MODIFICATIONS

ANNEX

CONTENTS

ILLUSTRATIONS

152

1. INTRODUCTION

This annex reports the equipment installation work tasks to be performed on the whirl site in support of research and development testing of a twin-engine helicopter hot cycle rotor and propulsion system for use on a research vehicle.

1.1   The work tasks are logically classified into two categories:

      a. Removal of existing items

      b. Installation of new items on existing or reworked hardware.

These classifications are discussed in detail in the body of this report.

1.2   Figure 1.2-1 shows the single engine rotor unit during whirl testing for the feasibility demonstration program.

Figure 1.2-1. Hot Cycle Rotor During Whirl Test.

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2. REMOVAL OF EXISTING EQUIPMENT

A listing of items to be removed from the whirl site is presented below:

a. J-57 engine

b. J-57 engine support and support base

c. Hot gas ducts

d. Butterfly valves

e. Hot gas duct supports

f. Upper work platform

g. Work scaffold support posts

h. Parts of the lowerable work platform

i. Control van pilot and flight engineer panels

j. Miscellaneous items associated with items a. through i. above.

3. INSTALLATION OF EQUIPMENT

Preliminary layout of the whirl tower facility includes all of the integrated equipment required at the whirl test site. This facility will function as the center of development testing for a twin-engine hot cycle rotor and propulsion system to power a helicopter research vehicle. The following pages discuss the integral parts.

3.1   INSTALLATION SEQUENCE

The equipment installation sequence of the whirl site will follow the removal of the items in Section 2.

The installation sequence will be dictated by the planned test program. It is of course desirable that all the test components simulate as closely as possible the detail design that will be installed on the research aircraft. However, the short elapsed time from contract go-ahead to start of whirl test precludes the availability of 100-percent flight hardware for the whirl test. Accordingly, in order to accelerate the overall program, the rotor-propulsion system will be brought gradually up to flight configuration during the course of testing, with the test program arranged to match this gradual build-up. Instrumentation, likewise, will be installed progressively as required by each test. For instance, the test program includes initial tests of the two T64 engines exhausting into a common duct without the rotor. To satisfy this requirement, engines and ducts will be installed, and engine and duct tests will be performed. However, for this phase of the tests, all engine cowlings and fairings will be omitted, and only partial installation of recording equipment will be required since rotor data will not be measured. Engine calibration and sound measurements are other examples of items which permit partial instrumentation. Further discussion on installation build-up sequence is given in the following sections of this report.

3.2   ENGINE SUPPORT STRUCTURE

This structure will support the engines, ducts, diverter valves, and tailpipes. It will simulate the research vehicle as nearly as possible with respect to nacelles, pylons, and cowlings. Hard points on the tower structure proper and the yaw control supports will by necessity differ from the flight vehicle. A preliminary drawing is presented in Figure 3.2-1.

3.2.1   Ducting

Initial tests will be made of the ducts between the engine and rotating seal to determine flow profiles and to obtain a preliminary

check of the dynamic behavior of twin engines exhausting into a common duct. The ducts will be made from Inconel "X" by conventional forming and welding methods.

The routing of the ducts is shown in Figure 3.2-1.

3.2.2   Diverter Valves

The diverter valves will be identical to those of the research vehicle. These valves are discussed in depth in the preceding Preliminary Design Report, in Paragraphs 3.4 and 9.2.2.

3.2.3   Actuating Systems

The present rotor actuators will be used during the initial engine rotor combination check-out. Flight-type actuators will be installed at approximately the fifteenth hour of accumulated whirl time. These actuators are discussed in Paragraphs 9.4.1 and 9.6 of the preceding Preliminary Design Report.

Actuator servos and engine power quadrants, and yaw control valves will be operated by cable linkage. Blade tip cascade valves and diverter valves will be operated by electric actuators and hydraulic pistons respectively.

3.3   REMOTE CONTROLS

Remote controls are required for the engine power quadrants, cascade valves, diverter valves, rotor actuators, and yaw control valves.

3.3.1   Cable Controls

Hand-operated cable controls will be utilized to control the positions of the engine power quadrants and rotor control actuators. The rotor control actuator cables will be linked to a collective and cyclic mixer.

3.3.2   Cable Linkage

Footpedal-operated cable linkage will position the yaw control valves.

3.3.3   Switch

A selective three-position switch will be used to select diverter valve position through use of a three-position solenoid actuated piston.
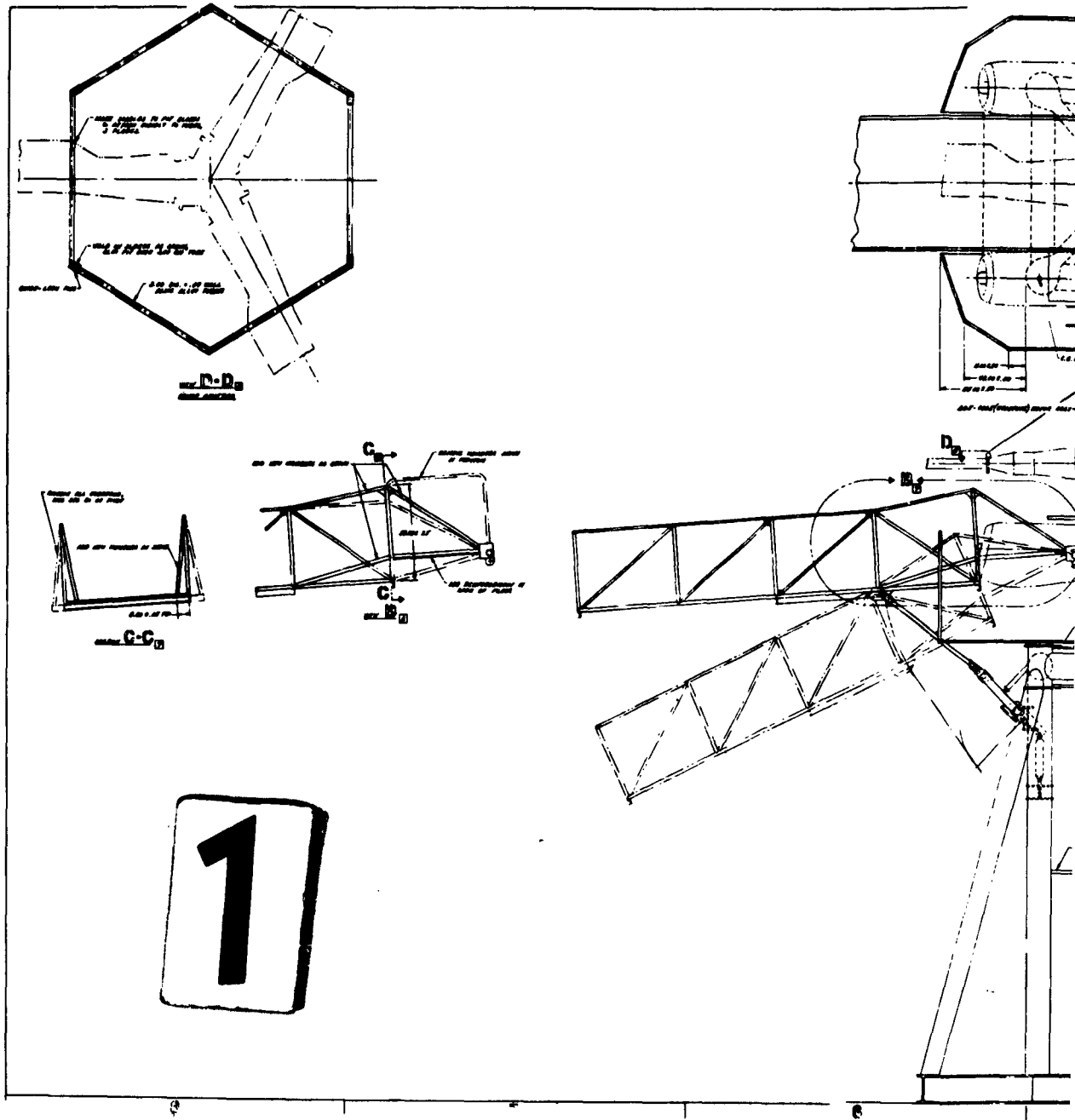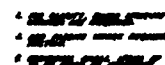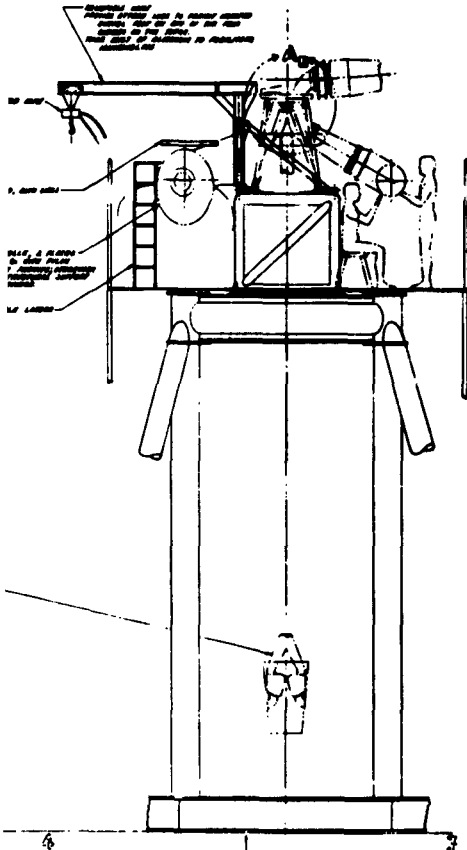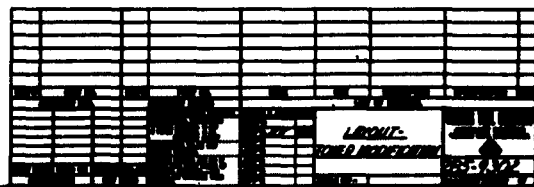
Figure 3.2-1. Layout-Tower Modification.

159

3.3.4   A single-pole, double-throw switch will be used to close or open the cascade valves.

3.4   CONTROL ROOM

The additional instrumentation for panel presentations and permanent recordings of test data establishes a need for more floor space within the control room. The control room will be extended 6 feet in length. It will be built from the same type of materials as the existing control room. This addition is shown in Figure 3.4-1.

The existing instrument panel will be removed and a new one fabricated to accommodate that instrumentation called out in Figure 3.4-2 for the pilot and flight engineer.

3.5   FUEL SYSTEM

The fuel system will be identical to that used for the J-57 during the feasibility demonstration except for changes associated with two engines instead of one.

3.6   GOVERNING

Governing will be accomplished by the T64 fuel control. The power turbine signal input will be accomplished by a hydraulic pump driven by the rotor accessory gear box coupled to a hydraulic motor mounted on the fuel control to sense the speed of the removed power turbine. This is discussed in Paragraph 9.2.4 of the preceding Preliminary Design Report.

3.7   HYDRAULIC SYSTEM

Hydraulic power will be derived from two independent sources, i.e., the T64 engines and the rotor accessory drive gear box. The power will be utilized for actuation of the diverter valves and rotor actuators.

3.8   ELECTRICAL SYSTEM

The electrical system will consist of conventional utilization of slip rings, transducers, solenoids, ignitors, recorders, etc.

An additional 100-ampere, 220-volt, 60-cps power line will be installed at the whirl site to accommodate the additional instrumentation. Also to be added is a 120-volt, 60-cps/24-volt dc converter to supply power to two CEC recorders.

3.9   STARTING SYSTEM

The starting system will be similar to that used for the J-57 during the feasibility demonstration of the hot cycle rotor.

Air impingement at the turbine will be used instead of a pneumatic torque converter but the same MA-1 compressor will supply the compressed air. A solenoid-operated air valve will be installed in the air supply line to each engine and be remotely controlled from the control van by a double-throw, four-pole toggle switch. This will provide the capability of selective single engine starting procedure and power-on panel indication.

3.10   INLET NOISE SUPPRESSOR

The T64 engine inlets will be facing away from the neighboring Loyola University. The noise level generated by the T64 engines is not expected to disrupt normal activities or to originate any complaints about noise. The directional characteristics of turbine inlet noise excludes it as a problem to personnel in the control van, surrounding areas, or Loyola University.

The noise level at different frequency bands and locations will be determined immediately as an initial phase of rotor and engine installation and operation check-out. A decision will be made following this determination if silencing is required and the size and type of inlet silencer to be installed. A preliminary installation is shown in Figure 3.10-1.

3.11   INSTRUMENTATION

The instrumentation requirements are increased considerably by the change of emphasis from a feasibility demonstration to an engine and rotor compatibility development directed toward actual flight of a research vehicle.

The engine performance during the feasibility demonstration was of minor importance and a minimum number of sensors were employed to ascertain its proper operation. With an additional engine to instrument and increased interest on engine performance in regard to transient conditions, the requirement of permanent performance recordings is increased. An additional 50-channel, galvanometer-type recorder and an additional potentiometer-type, 144-point recorder are planned for setup. Flash timing will be added and used to synchronize the timing lines on the galvanometer type recorders. Engine temperature and pressure traces will be placed as far as practicable adjacent to the dependent rotor temperature and pressure traces on the same recorder to simplify the determination of interrelationships.
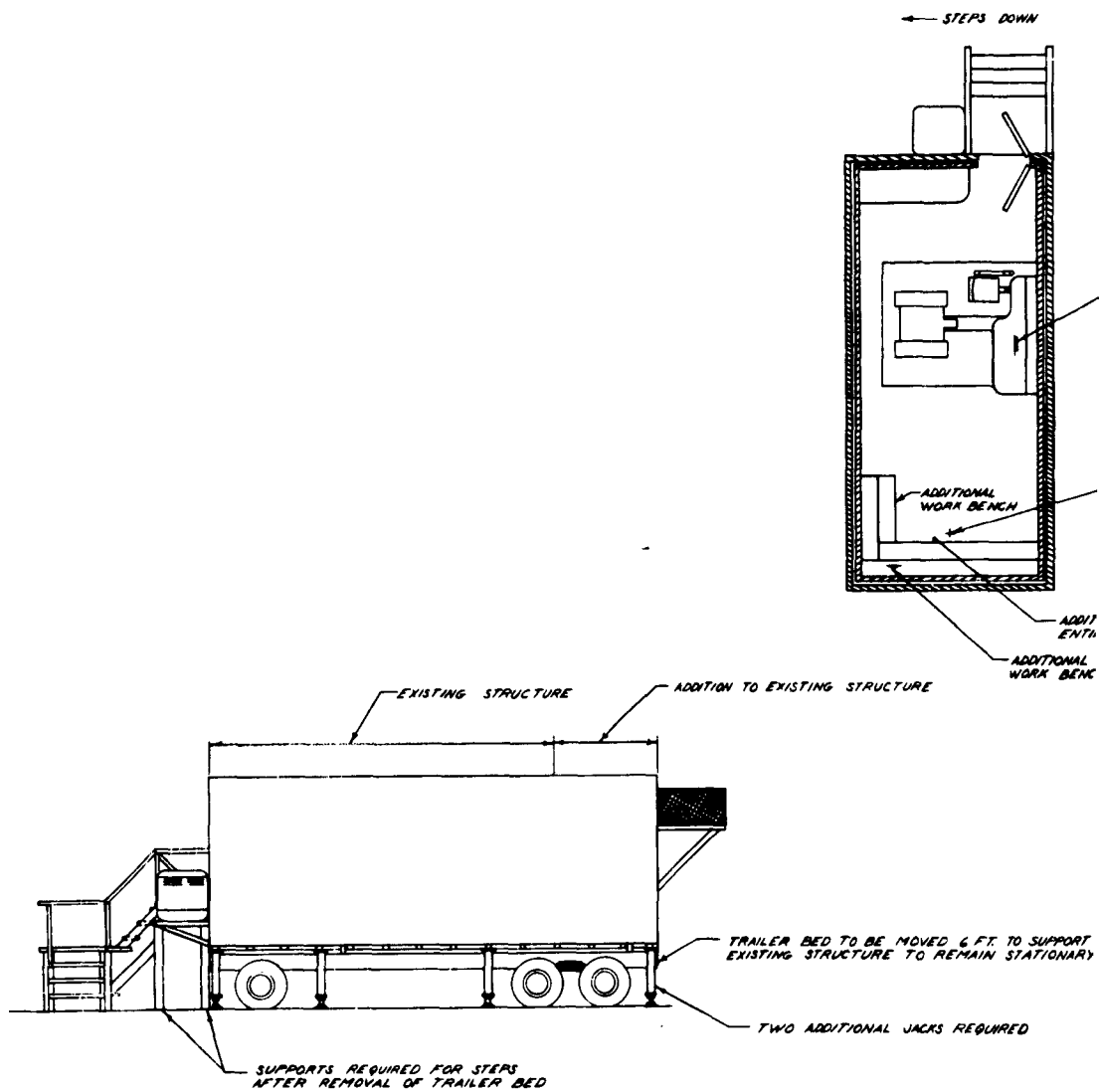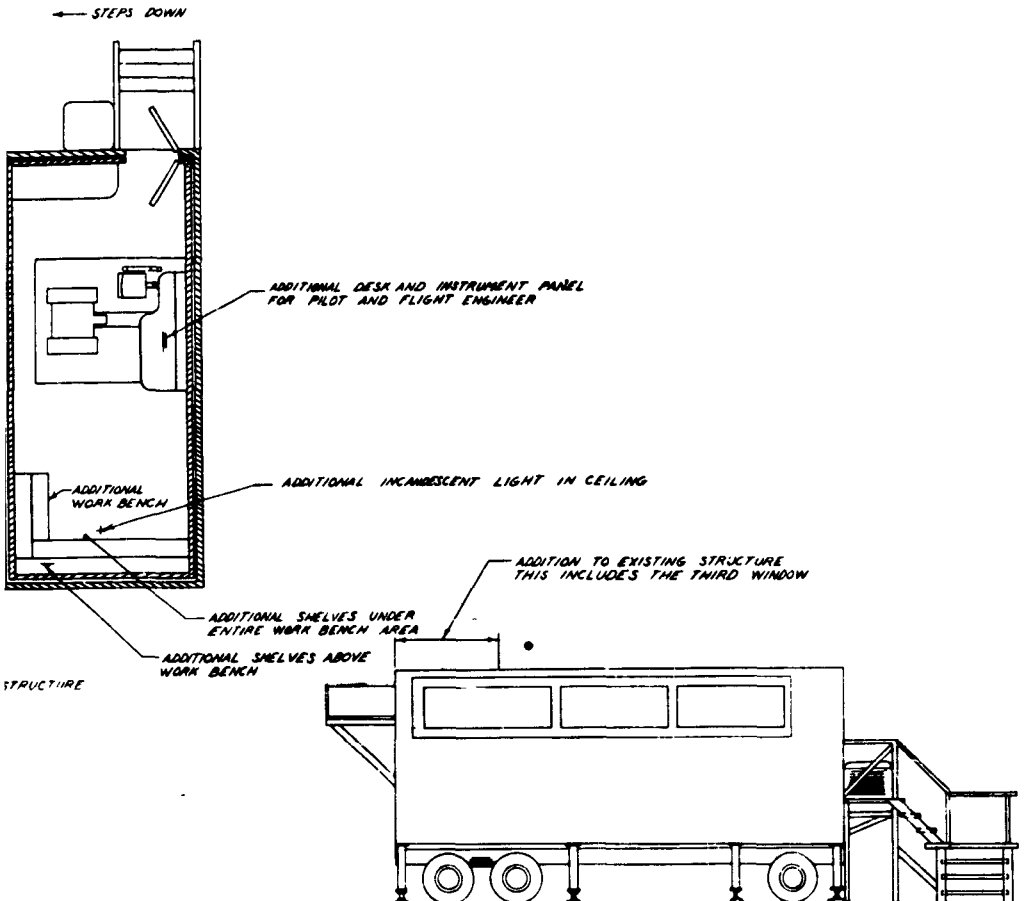
162

STEPS DOWN

ADDITIONAL
WORK BENCH

ADDIT
ENTI

ADDITIONAL
WORK BENC

EXISTING STRUCTURE

ADDITION TO EXISTING STRUCTURE

TRAILER BED TO BE MOVED 6 FT. TO SUPPORT
EXISTING STRUCTURE TO REMAIN STATIONARY

TWO ADDITIONAL JACKS REQUIRED

SUPPORTS REQUIRED FOR STEPS
AFTER REMOVAL OF TRAILER BED

Figure 3.4-1. Control Van Modifications-Whirl Test.

163

← STEPS DOWN

ADDITIONAL DESK AND INSTRUMENT PANEL
FOR PILOT AND FLIGHT ENGINEER

ADDITIONAL INCANDESCENT LIGHT IN CEILING

ADDITIONAL
WORK BENCH

ADDITION TO EXISTING STRUCTURE
THIS INCLUDES THE THIRD WINDOW

ADDITIONAL SHELVES UNDER
ENTIRE WORK BENCH AREA

ADDITIONAL SHELVES ABOVE
WORK BENCH

STRUCTURE

285-9304

TO BE MOVED 6 FT. TO SUPPORT ADDITIONAL STRUCTURE
TURE TO REMAIN STATIONARY

IONAL JACKS REQUIRED

GENERAL NOTES:

1. MAKE FROM THE SAME KIND OF MATERIALS AS IN
THE EXISTING STRUCTURE DRAWING 285-074.
2. INSTRUMENTS ARE CALLED OUT ON DRAWING
285-

REQD	PART NO.	REQD	PART NO.	NAME	SIZE	DESCRIPTION	SPECIFICATION
ASSEMBLY OPP.		ASSEMBLY SHOWN		LIST OF MATERIAL			
						CONTROL VAN MODIFICATIONS	
						- WHIRL TEST -	
NEXT ASSY	USED ON						285-9304
APPLICATION		QTY REQD					CODE 02731

HUGHES TOOL COMPANY
AIRCRAFT DIVISION
CULVER CITY, CALIFORNIA

SCALE

2

Figure 3.4-2. Engine and Rotor Instrumentation-Whirl Test.

165

TROL ROOM

HNICIANS

| CEC OSCILLOGRAPH RECORDER | —(15) |

| BROWN RECORDER | —(16) |

| PRESSURE INDICATOR | —(17) |

SEE NOTE 4

SEE NOTES 1 AND 2

SEE NOTE 3

285-9303
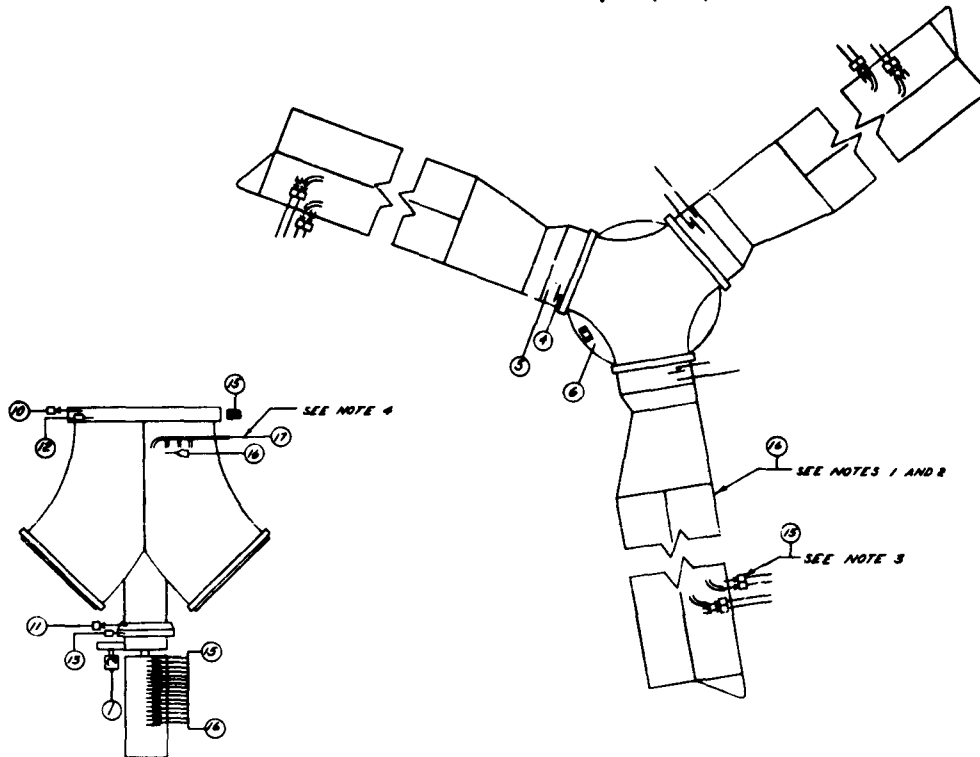
GENERAL NOTES

1. THERMOCOUPLE INSTALLATIONS FOR THE ROTOR ARE CALLED OUT ON DRAWING 285-1008.
2. STRAIN GAGE BRIDGE LOCATIONS ARE CALLED OUT ON DRAWING 285-1007.
3. ROTOR DUCT PRESSURE TRANSDUCERS WILL REQUIRE DEVELOPMENT TO OPERATE AT CENTRIFUGAL LOADING OF 585 G's AND ELEVATED TEMPERATURE TO 1225 °F.
4. STATIONARY DUCT PRESSURE PROBES AND THERMOCOUPLES ARE CALLED OUT ON DRAWING 285-1015

REQD	PART NO.	REQD	PART NO.	NAME	SIZE	DESCRIPTION	SPECIFICATION
	ASSEMBLY OPP.		ASSEMBLY SHOWN			LIST OF MATERIAL	

DRWN J Davis
CHK'D
APP'D
APP'D
APP'D
APP'D
APP'D
APP'D

ENGINE AND ROTOR INSTRUMENTATION - WHIRL TEST

HUGHES TOOL COMPANY
AIRCRAFT DIVISION
CULVER CITY, CALIFORNIA

285-9303

| NEXT ASSY | USED ON | | | | |
| APPLICATION | | QTY REQD | | | |

SCALE NONE

CODE 02731 SHEET OF

2

Figure 3.10-1. Whirl Site Assembly-Hot Cycle Research Program.

167

460 FT. TO
EDGE OF RUNWAY (REF)

ILLAGE DAM
ROUND TRAILER

VEL TRAILER

WORK STAND

INLET NOISE SUPPRESSOR (SPACE PROVISIONS ONLY)
SIZE AND TYPE TBD

INCREASE ELECTRICAL POWER
SUPPLY AS REQUIRED

ADD NEW 2" GAS LINE
FROM PLANT

285-0700 WHIRL TOWER ASSY
MODIFIED PER 385-9302

164 FT.

385-9304 MODIFIED CONTROL VAN

CHAIN LINK FENCE

200 FT. TO
EDGE OF ROADWAY (REF)

ogram.

2

_ = 5 - 9300

REQD	PART NO.	REQD	PART NO.	NAME	SIZE	DESCRIPTION	SPECIFICATION
	ASSEMBLY OPP.		ASSEMBLY SHOWN			LIST OF MATERIAL	

DRWN
CHK'D
APP'D
APP'D
APP'D
APP'D
APP'D

NEXT ASSY | USED ON
APPLICATION | QTY REQD

WHIRL SITE ASSY-
HOT CYCLE
RESEARCH PROGRAM

HUGHES TOOL COMPANY
AIRCRAFT DIVISION
CULVER CITY, CALIFORNIA

285 - 9300

SCALE 1/100

CODE 02731 SHEET OF

Pressure transducers, total and static, are to be installed at the rotor tips in each duct to acquire further knowledge of the gas conditions prior to exiting at the cascades. Present thinking consists of three possible solutions:

a. Conventional transducers mounted near the hub with probes running the length of the blade.

b. Conventional transducers mounted near the rotor tips with short probes running into the ducts.

c. A new bondable miniature transducer encapsulated and mounted on the airfoil with short probes running into the ducts. Proper operation of either of these methods is expected to require development because of the elevated temperatures and high centrifugal forces.

Additional thermocouples will be mounted in the air spaces surrounding the engine and its components to ascertain that the components have a proper operating environment compatible with their specification limits.

Additional instrumentation will be installed to read-out engine oil temperature, generator speed, turbine exit temperature and pressure, inlet air temperature and pressure, etc. Most of these will be recorded permanently as well as being displayed on either the pilot's or flight engineer's panel.

Cascade valve position, crack wire, and hub tilt stop indicators will be installed and wired through a slip ring to give a visual indication on the pilot's panel.

A preliminary drawing of the instrumentation with pilot's and flight engineer's instrument panel requirements is shown in Figure 3.4-2.

3.12 COOLING SYSTEM

The T64 engine will be installed on the whirl tower without cowlings for the initial phase of testing. This will permit visual inspection of engine fuel and oil fittings and external components during engine run-up. There will be sufficient air circulation to allow running within temperature limit specifications.

Engine cowlings will be installed at approximately the 15-hour whirl test time accumulation. These cowlings will be of the same design as used on the research vehicle. Recessed louvers will be formed in the cowlings to admit cooling outside air. Provisions will be made to utilize, if necessary, 14-stage compressor bled air to induce pumping

•

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of nacelle air through air ejector which exhausts outboard. This ejector is shown in Figure 21 of the preceding Preliminary Design Report. Engine oil will be cooled by a fuel flow cooled heat exchanger mounted on the engine. Rotor oil will be cooled by an ambient air heat exchanger.

The fore and aft spars of the rotor blades will be cooled by the centrifugal pumping of air outboard through the passages formed by the leading edge and trailing edge fairings.

3.13   LUBRICATION SYSTEM

The lubrication system of both the engine and rotor are shown in schematic form in Figure 3.13-1. Provisions for pumping rotor lubrication oil will be made within the rotor accessory drive gear box. The engine oil will be pumped by its own components accessory drive.

3.14   MAINTENANCE SUPPORT EQUIPMENT

The removal of items f, g, and h in Section 2 to make space for the installation of two T64 engines, diverter valves, and ducts deletes the work platforms required for maintenance and inspection. Therefore, it is planned to install a removable hoist, an engine work platform, a rotor work platform and access stairways, and to rework the lowerable blade work platform. Rented compressor equipment during the feasibility demonstration proved inadequate; therefore, a 2-inch compressed air supply line will be installed.
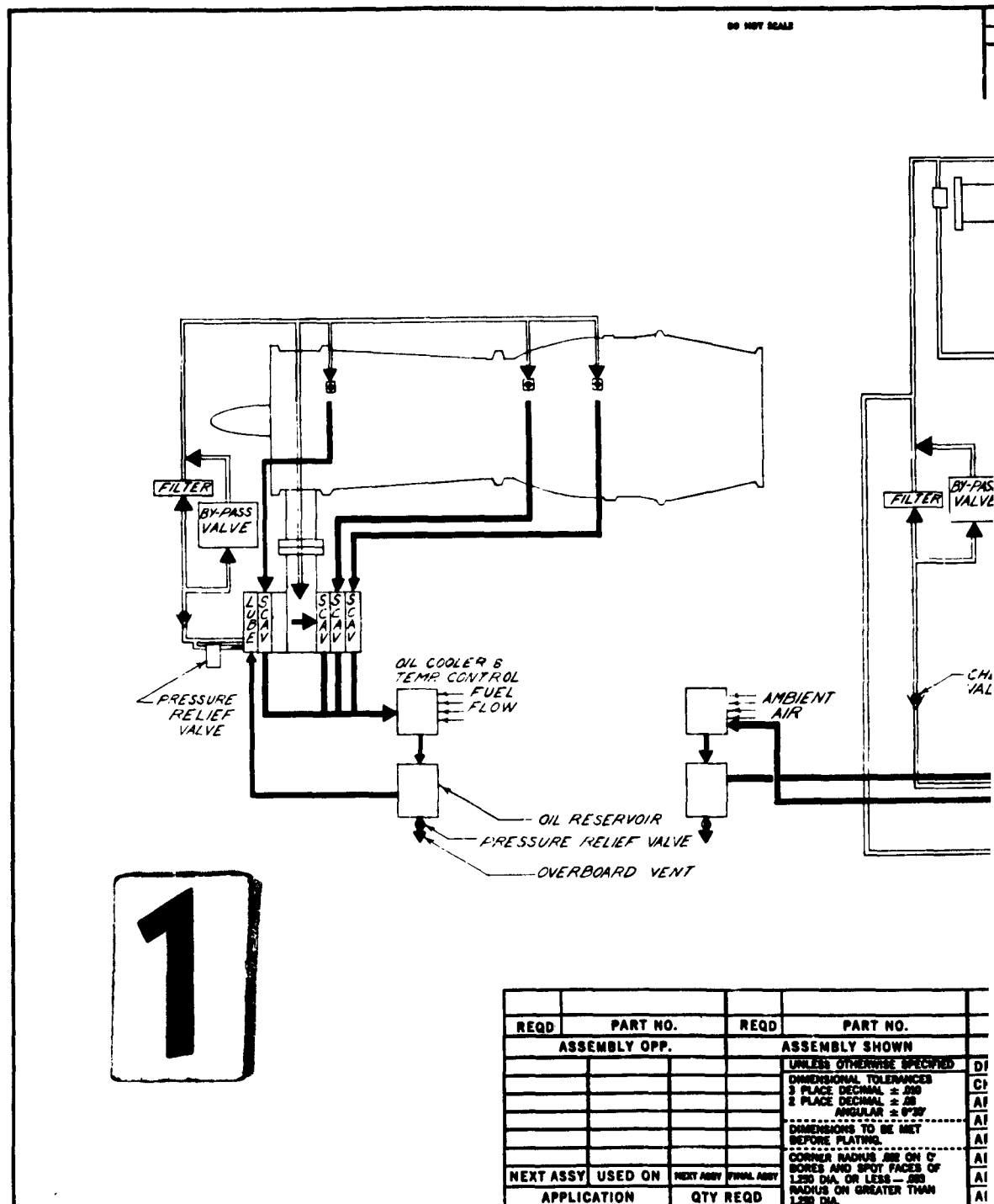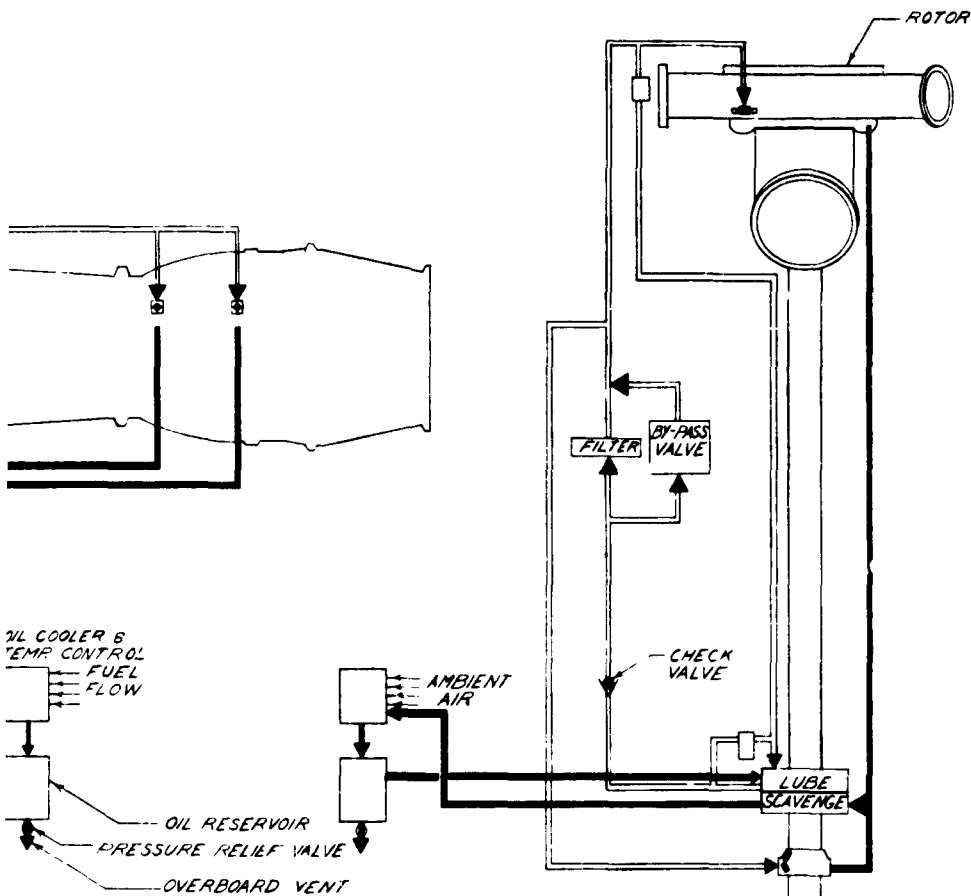
Figure 3.13-1.   Engine and Rotor Lubrication Systems-Schematic.

171

— ROTOR

2

9305

BY-PASS
FILTER VALVE

— CHECK
VALVE

OIL COOLER &
TEMP. CONTROL
— FUEL
FLOW

AMBIENT
AIR

LUBE
SCAVENGE

- - OIL RESERVOIR
— PRESSURE RELIEF VALVE
—OVERBOARD VENT

▬▬SCAVENGE & LOW PRESSURE OIL
═══ HIGH PRESSURE OIL

REQD	PART NO.	REQD	PART NO.	NAME	SIZE	DESCRIPTION	SPECIFICATION
ASSEMBLY OPP.		ASSEMBLY SHOWN		LIST OF MATERIAL			
			UNLESS OTHERWISE SPECIFIED	DRWN			
			DIMENSIONAL TOLERANCES 3 PLACE DECIMAL ± .010 2 PLACE DECIMAL ± .03 ANGULAR ± 0°30'	CHK'D		ENGINE AND ROTOR	HUGHES TOOL COMPANY
				APP'D		LUBRICATION SYSTEMS -	AIRCRAFT DIVISION CULVER CITY, CALIFORNIA
			DIMENSIONS TO BE MET BEFORE PLATING.	APP'D		SCHEMATIC	
			CORNER RADIUS .062 ON C' BORES AND SPOT FACES OF	APP'D			385 -9305
NEXT ASSY	USED ON	NEXT ASSY FINAL ASSY	1.250 DIA. OR LESS — .063 RADIUS ON GREATER THAN	APP'D			
APPLICATION		QTY REQD	1.250 DIA.	APP'D		SCALE	CODE 02731 SHEET OF

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First US Army	3
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Sixth US Army	1
USAIC	2
USACGSC	1
USAWC	1
USAATBD	1
USAARMBD	1
USAAVNBD	1
USATMC(FTZAT),ATO	1
USAPRDC	1
DCSLOG	2
Rsch Anal Corp	1
ARO, Durham	2
OCRD, DA	1
USATMC Nav Coord Ofc	1
NATC	2
CRD, Earth Scn Div	1
USAAVNS, CDO	1
DCSOPS	1
OrdBd	1
USAQMCDA	1
QMFSA	1
CECDA	1
USATCDA	1
USATB	1
USATMC	20
USATC&FE	4
USATSCH	3
USATRECOM	17
USATTCA	1
USA Tri-Ser Proj Off	1
TCLO, USAABELCTBD	1
USASRDL LO, USCONARC	2
USATTCP	1
OUSARMA	1
USATRECOM LO, USARDG(EUR)	1

USAEWES	1
TCLO, USAAVNS	1
USATDS	5
USARPAC	1
EUSA	1
USARYIS/IX CORPS	2
USATAJ	6
USARHAW	3
ALFSEE	2
USACOMZEUR	3
USARCARIB	4
AFSC (SCS-3)	1
APGC (PGAPI)	1
Air Univ Lib	1
AFSC	2
ASD (ASRMPT)	1
CNO	1
ONR	3
BUWEPS, DN	5
ACRD(OW), DN	1
BUY&D, DN	1
USNPGSCH	1
CMC	1
MCLFDC	1
MCEC	1
MCLO, USATSCH	1
USCG	1
NAFEC	3
Langley Rsch Cen, NASA	2
Geo C. Marshall Sp Flt Cen, NASA	1
MSC, NASA	1
Ames Rsch Cen, NASA	2
Lewis Rsch Cen, NASA	1
Sci & Tech Info Fac	1
USGPO	1
ASTIA	10
HumRRO	2
US Patent Ofc, Scn Lib	1
ASD, FCL	1
USAMOCOM	3
USSTRICOM	1
USAMC	1
Hughes Tool Co	10

174

Hughes Tool Company - Aircraft
Division, Culver City, California,
HOT CYCLE RESEARCH VEHICLE,
TCREC Technical Rept 62-102,
March 1963, 179 pp. (Contract DA
44-177-TC-832) USATRECOM Task
9R38-01-020-03.

Unclassified Report

A preliminary design study has been
completed of a twin-engine hot cycle
research helicopter utilizing the
(over)

1. Hot Cycle Re-
search Vehicle
2. Contract
DA 44-177-
TC-832

Hughes Tool Company - Aircraft
Division, Culver City, California,
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TCREC Technical Rept 62-102,
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research helicopter utilizing the
(over)

1. Hot Cycle Re-
search Vehicle
2. Contract
DA 44-177-
TC-832

existing rotor with the following design objectives:

1. 15,300 pounds design gross weight
2. 25,500 pounds alternate heavy-lift hovering gross weight.
3. 145 knots at a gross weight of 15,300 pounds, helicopter mode.
4. 197 knots at a gross weight of 10,000 pounds, autogyro mode.

Major design areas covered in the study include the rotor modifications, airframe, engine installation, engine controls, flight controls, diverter valves, blade duct valves, flight instrumentation, electrical system, hydraulic system, and fuel system.